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RESEARCH MEMORANDUM

ALTITUDE PERFORMANCE INVESTIGATION OF TWO SINGLE-ANNULAR TYPE COMBUSTORS AND THE PROTOTYPE J40-WE-8 TURBOJET ENGINE COMBUSTOR WITH VARIOUS COMBUSTOR-INLET AIR PRESSURE PROFILES

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ALTITUDE PERFORMANCE INVESTIGATION OF TWO SINGLE-ANNULAR TYPE COMBUSTORS AND THE PROTOTYPE J40-WE-8 TURBOJET ENGINE COMBUSTOR WITH VARIOUS COMBUSTOR INLET-AIR PRESSURE PROFILES

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SUMMARY

Data were obtained for three single-annular type combustors with different combustor inlet-air pressure profiles over a range of engine speeds at an altitude of 30,000 feet and a flight Mach number of 0.62. The combustors with a lower percentage of total hole area at the inner wall had a higher combustor-outlet temperature profile near the inner wall than the combustor with equal hole-area distributions; the converse was true near the outer wall. As the combustor inlet-air pressure profile was lowered (corresponding to a reduction in air flow) at the inner portion of the passage height, the combustor-outlet temperature profile near the inner wall was raised. Similar trends were encountered near the outer wall. Combustor pressure-loss coefficient was not affected by hole-area distribution but was affected by total hole area and inlet-air pressure profile. For combustors with total hole areas of 877 and 809 square inches, the pressure-loss coefficients were 10.8 and 12.4, respectively, at a combustor density ratio of 2.2. For changes in inlet-air pressure profile, the pressure-loss coefficient varied from 10.8 to 15.8, at a density ratio of 2.2. There was no discernible effect of the aforementioned variables on combustion efficiency.

Combustor performance data were also obtained with the compressor-combustor configuration of the turbojet engine designated the prototype J40-WE-8. These data were obtained over a range of altitudes from 15,000 to 55,000 feet and flight Mach numbers from 0.17 to 0.99. For the prototype J40-WE-8 turbojet-engine combustor, combustion efficiency at a corrected engine speed of 7600 rpm decreased from 0.98 at an altitude of 15,000 feet to 0.83 at an altitude of 55,000 feet at a flight Mach number of 0.62 and open exhaust-nozzle area (area of 534 sq in.).

A good correlation was obtained when combustion efficiency was presented as a function of a combustion parameter and engine fuel-air ratio. These data indicated that at values of combustion parameter below 34,000 pounds-°R-second per cubic foot there was a fuel-air ratio that resulted in an optimum combustion efficiency for a given value of combustion parameter.

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INTRODUCTION

An investigation of the performance of the XJ40-WE-6 turbojet engine in the NACA Lewis altitude wind tunnel disclosed that the engine operated with compressor surge and a combustor-outlet temperature inversion within the desired operating speed range. As a result of changes made in the setting of the blades in the compressor and a study of the configuration of the combustor, conducted in cooperation with the engine manufacturer, the compressor surge was displaced out of the operating speed range and the combustor-outlet temperature inversion was corrected. These results are reported in references 1 and 2.

In correcting the combustor-outlet temperature inversion, three single-annular-type combustors having slightly different air-passage geometry were evaluated on the engine. Correcting the compressor surge by making changes to the blade settings resulted in different inlet-air pressure profiles at the inlet to the combustors and made possible a determination of the effect of inlet-air pressure profile on combustor performance. This investigation was conducted over a range of engine speeds, at an altitude of 30,000 feet, and a flight Mach number of 0.65.

The XJ40-WE-6 engine having the improved compressor and combustion-chamber configuration was designated the prototype J40-WE-8 turbojet engine without an afterburner. Combustor performance data on the prototype J40-WE-8 engine were obtained over a range of altitudes from 15,000 to 55,000 feet, flight Mach numbers from 0.17 to 0.99, and over a range of engine speeds at five fixed exhaust-nozzle areas. These combustor data constituted the first evaluation in an altitude facility of the performance of a single-annular combustor with spring-loaded variable-area fuel nozzles operating as an integral component of a turbojet engine.

Combustor data are presented herein to show the correlation of combustion efficiency with engine fuel-air ratio and a combustion parameter expressed in terms of inlet variables P_4T_4/V_b. (All symbols used in this report are given in appendix A.)

The performance of the prototype J40-WE-8 turbojet-engine combustor and three other different types of combustors are compared herein by data which shows the variation of combustion efficiency with fuel-air ratio and combustion parameter P_4T_4/V_b for the different combustors.

APPARATUS

Engine

The turbojet engine used at the start of this investigation was designated the XJ40-WE-6. Subsequent compressor and combustor configurations resulted in the prototype J40-WE-8 turbojet engine without afterburner (fig. 1). A manufacturer's rating for the prototype J40-WE-8 turbojet engine is not available at the present time; however, its rating would be similar to the rating of the XJ40-WE-6 turbojet engine, which had a static sea-level thrust of 7500 pounds at an engine speed of 7260 rpm and a turbine-inlet temperature of 1425° F (1885° R). At this operating condition the air flow was approximately 142 pounds per second, and the combustor-inlet total pressure, total temperature, and velocity (based on the maximum cross-sectional area of the combustor, 6.40 sq ft) were 10,600 pounds per square foot absolute, 870° R, and 101 feet per second, respectively. The principal components of the engine were an eleven-stage axial-flow compressor, single-annular combustor, two-stage turbine, diffuser, and variable-area exhaust nozzle.

A number of different compressor configurations were obtained in the compressor development program, and data were selected for presentation herein from three configurations. These configurations, which were designated compressors 1 to 3, were chosen because they provided a wide range of combustor inlet-air pressure profiles.

Combustors

Combustion data were obtained with three combustors (supplied by the manufacturer) which were of the single-annular type, differing only in the perforations in the inner and outer walls of the combustor basket and in some mechanical strengthening features. These combustors had a maximum cross-sectional area of 6.40 square feet. The combustors, designated A, B, and C, are shown in figures 2, 3, and 4, respectively. A cross section of the combustors and a developed sketch of an element of surface from the combustor baskets for each of the three combustors are shown in figure 5. The variation of total hole area with combustor length for the three combustors is presented in figure 6. The total hole area includes the area of the

openings shown in figure 5 and the various circumferential openings located at the inner and outer walls of the combustor. As shown in figure 6, the total hole area for combustors A and C was 796 and 877 square inches, respectively; however, approximately the same percentage of hole area was provided at the inner and outer walls of the combustor basket. The distribution of total hole area was 46.5 percent at the inner wall and 53.5 percent at the outer wall. Combustor B, which had a total hole area approximately the same as combustor A, had equal area distribution at the inner and outer walls of the combustor basket.

The splitter (fig. 5) divided the air flow entering the combustor into two annular passages formed by the combustor basket and the inner and outer walls of the combustor. Engine fuel was admitted and sprayed downstream in the combustor through 16 spring-loaded variable-area nozzles located at the upstream end of the combustor. Through the combined action of an engine-fuel distributor, equalizing valves, and spring-loaded variable-area nozzles, the fuel flow through each of the 16 nozzles was maintained equal at all fuel flows.

INSTALLATION AND INSTRUMENTATION

The engine was mounted on a wing section that spanned the 20-foot-diameter test section of the altitude wind tunnel (fig. 1). Dry refrigerated air was supplied to the engine from the tunnel make-up air system through a duct connected to the engine inlet. Throttle valves were installed in the duct to permit regulation of the pressure at the inlet of the engine. Instrumentation for measuring pressures and temperatures was installed at various stations in the engine (figs. 7 and 8). Ten sonic probe thermocouples, which could be traversed radially, were used at the combustor-outlet station (fig. 8(c)) to obtain temperature profiles.

PROCEDURE

Dry refrigerated air was supplied to the engine at the standard temperature for each flight condition with the exception that the minimum temperature obtained was about -20° F (440° R). The air, at approximately sea-level pressure at the entrance of the make-up air system, was throttled to a total pressure at the engine inlet corresponding to the desired flight condition, with complete free-stream ram pressure recovery assumed.

Combustor performance data, showing the effect of different combustor inlet-air pressure profiles and combustor hole-area

distribution on combustor performance, were obtained at an altitude of 30,000 feet, a flight Mach number of 0.62, and over a range of engine speeds.

The combustor of the prototype J40-WE-8 turbojet engine, which consisted of compressor 1 and combustor A, was investigated over a range of altitudes from 15,000 to 55,000 feet, flight Mach numbers from 0.17 to 0.99, at several constant exhaust-nozzle areas, and over a range of engine speeds.

Complete radial surveys of the combustor-outlet temperature using the sonic probe thermocouples were obtained at rated speed only. The engine fuel used was MIL-F-5624 at a temperature of about 80° F. This fuel had a lower heating value of 18,700 Btu per pound and a hydrogen to carbon ratio of 0.171. The methods of calculation are presented in appendix B.

RESULTS AND DISCUSSION

Effect of Changing Combustor Inlet-Air Pressure Profile and Hole Geometry on Combustor Performance

The effects on combustor performance of inlet-air pressure profiles and combustor hole-area distribution are discussed in terms of (1) temperature profile at the combustor outlet, (2) pressure-loss characteristics, and (3) combustion efficiency.

Combustor-outlet temperature profiles. - The effect of different combustor configurations on combustor-outlet temperature profiles for operating conditions at high and low engine speeds is shown in figures 9 and 10. As mentioned previously, radial temperature surveys at the combustor outlet (station 5, fig. 8(c)) were obtained only at rated spped. It has been shown, however, that turbine-outlet temperature profiles (station 6, fig. 8(d)) are indicative of turbine-inlet or combustor-outlet temperature profiles; therefore, turbine-outlet temperature profiles are presented at reduced engine speeds (fig. 9(d) and 10(d)). In the comparison of the combustor configurations the combustor inlet-air pressure profiles (compressor outlet-air pressure profiles) are the same. Any change in combustor performance may therefore be attributed to the difference in the combustor hole geometry. Combustors A and B, which are compared in figure 9, have about the same total hole area, but different hole-area distribution. The percentage hole-area distribution at the inner wall for combustors A and B was 46.5 and 50 percent, respectively. As shown in figure 9, the combustor-outlet temperature distribution was

affected by variations in hole-area distribution. The effect of changes in hole-area distribution at the inner and outer walls was to cause a radial shift (due to a restriction or damming effect) in air flow in the region between the compressor outlet, where the combustor inlet-air pressure profiles were measured, and the splitter (fig. 5). The decrease in hole area at the inner wall for combustor A resulted in lower air flow and, therefore, high combustor-outlet temperatures near the inner wall. Conversely, combustor A had relatively lower combustor-outlet temperatures near the outer wall.

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The combustors compared in figure 10 differ both in hole-area distribution and total hole area. Combustor B had a total hole area of 809 square inches, 50 percent of which was located on the inner wall, and combustor C had a total hole area of 877 square inches, 46.5 percent of which was located on the inner wall. This lower percent of total hole area and air flow at the inner wall of combustor C resulted in higher combustor-outlet temperatures near the inner wall as shown in figures 10(b) and 10(d). The reverse was again true at the outer wall.

Although the changes in combustor-outlet temperature profile for the different combustors have been explained on the basis of total hole-area distribution at the inner and outer walls, the effect of changes in the axial hole distribution (figs. 5 and 6) is also an influencing factor. It was not possible, however, from the data available to account for the effect of changes in the axial hole distribution.

The effect of combustor inlet-air pressure profile on combustor-outlet temperature profile is shown in figure 11. The splitter located at the upstream end of the combustor (fig. 5) tends to direct the air flow in the inner 55 percent of the passage height towards the inner wall of the combustor and the remaining portion of the air flow towards the outer wall. As shown in figures 11(a) and 11(c) the shift in total-pressure distribution with change in compressor configuration resulted in a greater percentage of the total air flow for compressor 2 relative to compressor 3 to be directed towards the inner wall of the combustor. This effect resulted in lower combustor-outlet temperatures at the inner portion of the passage height and higher temperatures at the outer portion of the passage height for compressor 2 (figs. 11(b) and 11(d)). Thus, for the series of combustors investigated the combustor-outlet temperature profile was shown to be influenced by the combustor inlet-air pressure profile as well as by the changes in combustor hole-area distribution discussed previously.

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Pressure-loss characteristics. - The effect of combustor configurations and combustor inlet-air pressure profiles on combustor pressure-loss coefficient $(P_4 - P_5)/q_b$ is presented in figure 12. Although there is considerable scatter in the data, particularly at low total density ratios, curves were faired through the points with the aid of trends established from data for other configurations and from windmilling engine tests. Combustors A and B, compared in figure 12(a), have about the same total hole area but differ in hole-area distribution. As shown, in figure 12(a) there was no apparent difference in pressure loss between the two combustors. Combustors B and C, having different hole areas and hole distributions, are compared in figure 12(b). The pressure loss is greater for combustor B which had the smaller total hole area. At a constant value of combustor density ratio of 2.2, the pressure-loss coefficient was 10.8 and 12.4 for combustors C and B, respectively. The data show, therefore, that over the range of hole geometry investigated the pressure loss was independent of hole area distribution (fig. 12(a)) and dependent on the total hole area (fig. 12(b)).

The effect of combustor inlet-air pressure profile on combustor total-pressure-loss coefficient of combustor C is shown in figure 12(c). The pressure loss for the air-pressure profile of compressor 2 was greater than that obtained with compressor 3. At a density ratio of 2.2, the pressure-loss coefficient was 10.8 and 15.8 for air-pressure profiles of compressors 3 and 2, respectively. Since the temperature profiles shown in figures 11(b) and 11(d) indicate that compressor 2 directs a greater proportion of the air flow toward the combustor inner wall than compressor 3, and also that the combustor inner wall had a lower percentage of the total hole area than the outer wall, the pressure-loss coefficient would tend to be greater for the air-pressure profile of compressor 2. Thus, it is apparent that the pressure-loss coefficient is sensitive to combustor inlet-air pressure profile; however, it is not possible to determine precisely whether the increase in pressure-loss coefficient associated with compressor 2 was due entirely to the increase in losses in mixing and turbulence in the combustor basket or in diffusion loss from the combustor inlet (compressor outlet) to the combustor.

Combustion efficiency. - The effect of combustor configurations and combustor inlet-air pressure profiles on combustion efficiency is shown in figure 13. In order to enable a direct comparison of the different combustors and inlet-air pressure profiles irrespective of differences in inlet pressure, temperatures, or velocities, the combustion correlation parameter P_4T_4/V_b was used. This combustion parameter is derived in reference 3. As will be shown later, there was an additional effect of fuel-air ratio on combustion efficiency.

Inasmuch as the various configurations were investigated at the same flight conditions, and over the same range of engine speeds and exhaust-nozzle areas, the fuel-air ratios for each of the configurations were essentially the same for any given value of combustion parameter shown in figure 13. The data show that for the configurations and pressure profiles studied there was no effect of these variables on combustion efficiency. Combustion efficiency remained approximately constant at 0.98 for values of combustion parameter greater than 34,000 pounds-^{o}R-second per cubic foot, and decreased for values of combustion parameter below 34,000 pounds-^{o}R-second per cubic foot to 0.60 at a combustion parameter of 8400 pounds-^{o}R-second per cubic foot.

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Performance of the Prototype J40-WE-8 Turbojet-Engine Combustor

The results presented in the previous discussion were obtained during the early phase of the investigation which consisted of a compressor development and combustor evaluation program of the XJ40-WE-6 turbojet engine. From this part of the investigation, as mentioned previously, a configuration comprised of compressor 1 and combustor A was selected for the prototype J40-WE-8 turbojet engine. This configuration was chosen because of improved compressor surge characteristics, elimination of combustor-outlet temperature inversion (references 1 and 2), and satisfactory mechanical reliability of the combustor. A performance evaluation of this configuration was obtained over a wide range of flight and engine operating conditions and is presented in the following section. Most of the performance data are presented at an exhaust-nozzle area of 534 square inches (open nozzle). The trends of the data for all the exhaust-nozzle areas were similar, but the effects on the combustor performance were somewhat greater with the open exhaust-nozzle area. Data for all exhaust-nozzle areas are presented in table I.

Combustion efficiency. - The effects of corrected engine speed, altitude, flight Mach number, and exhaust-nozzle area on combustion efficiency are shown in figure 14. Although flight condition, engine speed, and exhaust-nozzle area are not basic combustor variables, the data in figure 14 are shown in order to illustrate the variation in performance of the combustor in an engine. The variations in combustion efficiency for a given combustor configuration are primarily due to changes in combustor-inlet pressure, temperature, velocity, and fuel-air ratio as will be discussed later. At a flight Mach number of 0.62 and exhaust-nozzle area of 534 square inches, combustion

efficiency decreased from 0.98 at 15,000 feet to 0.83 at 55,000 feet, at a corrected engine speed of 7600 rpm (fig. 14(a)). The effect of altitude on combustion efficiency becomes even more pronounced at the lower engine speeds. Although the variables, flight Mach number and exhaust nozzle area, also affect combustion efficiency, the effects are less pronounced than the altitude effect as shown in figures 14(b) and 14(c), respectively. At a corrected engine speed of 7600 rpm and at an altitude of 35,000 feet, (fig. 14(b)) a change in flight Mach number from 0.17 to 0.99, increased combustion efficiency from about 0.955 to 0.995. In figure 14(c), which shows the effect of exhaust-nozzle area on combustion efficiency at 35,000 feet and flight Mach number of 0.62, combustion efficiency increased from about 0.97 to 0.98 as the exhaust-nozzle area was reduced from 534 to 420 square inches at a corrected engine speed of 7600 rpm.

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Combustor pressure-loss characteristics. - Combustor pressure-loss characteristics are presented in terms of engine parameters in figure 15 and of combustor parameters in figure 16. In both figures the pressure-loss characteristics include the pressure loss due to (1) the diffusion process from the combustor inlet (compressor outlet) to the combustor basket, (2) mixing and turbulence in the combustor basket, and (3) momentum pressure loss associated with the burning process. For all flight conditions and exhaust-nozzle areas, the combustor total-pressure-loss ratio $(P_4 - P_5)/P_4$ decreased with increasing corrected engine speed above a corrected engine speed of about 6000 rpm (fig. 15). For example, at an altitude of 35,000 feet, flight Mach number of 0.62, and exhaust-nozzle area of 534 square inches, the combustor total-pressure-loss ratio decreased from 0.040 to 0.031 as corrected engine speed increased from 6000 to 7400 rpm (fig. 15). This reduction in pressure-loss ratio with increasing corrected engine speed may be attributed to a more favorable combustor inlet-air pressure profile resulting in a more efficient diffusion process. At a constant value of corrected engine speed, decreasing altitude (fig. 15(a)) or increasing flight Mach number (fig. 15(b)) or exhaust-nozzle area (fig. 15(c)), in general, resulted in an increasing pressure-loss ratio. For instance, at a corrected engine speed of 7000 rpm, altitude of 35,000 feet, and flight Mach number of 0.62, increasing exhaust-nozzle area from 367 to 534 square inches resulted in an increase of total-pressure-loss ratio from 0.024 to 0.036 (fig. 15(c)).

The combustor pressure-loss characteristics are presented in terms of fundamental combustor parameters in figure 16. The combustor total-pressure-loss coefficient increased as the combustor total-density ratio was increased from 1.0 to 1.9, reaching a maximum value of 9.2 at a density ratio of 1.9. For values of density ratios above 1.9, the pressure-loss coefficient tends to decrease. From theoretical considerations (reference 4), the pressure-loss coefficient should vary linearly with density ratio. Possible factors in the disagreement are that the

efficiency of the diffusion process, as well as the mixing and turbulent losses in the combustion, varied as the density ratio was changed.

Correlation of Combustion Efficiency with Engine Fuel-Air Ratio and Combustion Parameter

Because the process of combustion is complex and depends on many factors it is difficult, if not impossible, to determine a combustion parameter which correlates combustion efficiency for all flight and engine operating conditions. However, some of the primary variables affecting combustion efficiency are considered in the combustion parameter P_4T_4/V_b derived in reference 3. In order to obtain a satisfactory correlation of combustion efficiency with combustion parameter P_4T_4/V_b, an additional parameter, engine fuel-air ratio, was introduced. Combustion efficiency is presented in figure 17 as a function of these two combustion parameters for two of the compressor-combustor configurations investigated. The data of figure 17(a) were obtained at altitudes from 15,000 to 55,000 feet and flight Mach numbers from 0.17 to 0.99. The data of figure 17(b) represent a range of altitudes from 15,000 to 45,000 feet and flight Mach numbers from 0.17 to 0.62. Although scatter is present, particularly at low values of P_4T_4/V_b, the curves for several narrow ranges of fuel-air ratio provide a reasonably good correlation of the data. In general, the data in figures 17(a) and 17(b) exhibit about the same magnitudes and trends. In figures 17(a) and 17(b), combustor efficiency begins to decline for values of P_4T_4/V_b below 34,000 pounds-$^{\circ}$R-second per cubic foot. Below this value of P_4T_4/V_b, combustion efficiency was sensitive to fuel-air ratio, and above this value, fuel-air ratio had a negligible effect.

The data of figure 17 are presented in figure 18 with fuel-air ratio as the abscissa in order to show more clearly the effect of fuel-air ratio on combustion efficiency. Because sufficient data were not available to completely separate the variables, P_4T_4/V_b and fuel-air ratio, each of the curves presented in figure 18 is for a small range of P_4T_4/V_b. These data indicate that over these small ranges of P_4T_4/V_b there was an optimum value of fuel-air ratio for maximum combustion efficiency. For example, for a range of P_4T_4/V_b of 6500 to 7500 pounds-$^{\circ}$R-second per cubic foot, combustion efficiency varied from 0.50 to 0.67 as fuel-air ratio was increased from 0.0066 to 0.0112, and a further increase in fuel-air ratio from 0.0112 to 0.0156 decreased combustion efficiency from 0.675 to 0.55.

Combustion efficiency probably varied with fuel-air ratio at a constant value of combustion parameter because of local rich and lean fuel-air ratio regions in the primary zone of the combustor. These regions may also be influenced by the degree of fuel atomization. At the high values of fuel-air ratio, some of the local regions in the primary zone are probably excessively rich in fuel, and combustion was incomplete because of a lack of oxygen; whereas, at the lower values of fuel-air ratio, some of the local regions were too lean for efficient
2626 combustion.

Comparison of Several Combustors from Different Turbojet Engines

Performance of four different current combustors is compared in figure 19. Combustion efficiency is shown as a function of combustion parameter P_4T_4/V_b at three different levels of fuel-air ratio. Combustor A was the combustor used in the prototype J40-WE-8 turbojet engine. Data for combustor M were not available below a combustion parameter of 20,000 pounds-$^{\circ}$R-second per cubic foot.

Combustion efficiency of all combustors shown was affected somewhat by fuel-air ratio, probably because of the rich and lean combustion regions previously discussed. This effect of fuel-air ratio was greatest at low values of combustion parameter P_4T_4/V_b.

For the range of combustor operating conditions investigated, the performance of combustors A, M, and N was approximately the same. These combustors have fuel systems that provide good fuel atomization and distribution over a wide range of fuel flows. Combustor P had a lower combustion efficiency than combustors A, M, and N, especially at low values of combustion parameter and fuel-air ratio. The low combustion efficiencies experienced with combustor P are felt to be primarily a result of the fixed-area fuel nozzles which provide poor spray and penetration characteristics at low fuel flows. Of course, combustion efficiency is primarily a function of matching the fuel and air properly and not of fuel injection alone; nevertheless, for the combustors presented, combustion efficiency is concluded to be primarily dependent on the method of fuel injection rather than the type of combustor used.

SUMMARY OF RESULTS

1. The effect of combustor hole-area distribution and combustor inlet-air pressure profile on combustor performance was obtained over a range of engine speeds at an altitude of 30,000 feet and a flight Mach number of 0.62:

(a) The combustors with a lower percentage of total hole area at the inner wall had a higher combustor-outlet temperature profile near the inner wall than the combustor with equal hole-area distribution; the converse was true near the outer wall. As the combustor inlet-air pressure profile was lowered (corresponding to a reduction in air flow) at the inner portion of the passage height, the combustor-outlet temperature profile near the inner wall was raised. Similar trends were 2626
encountered near the outer wall.

(b) Combustor pressure-loss coefficient was not affected by hole-area distribution but was affected by total hole area and inlet-air pressure profile. For combustors with total hole area of 877 and 809 square inches, the pressure-loss coefficient was 10.8 and 12.4, respectively, at a combustor density ratio of 2.2. For changes in inlet-air pressure profile, the pressure-loss coefficient varied from 10.8 to 15.8 at a density ratio of 2.2. There was no discernible effect of these variables on combustion efficiency.

2. With compressor 1 and combustor A, which was the configuration designated the prototype J40-WE-8, data were obtained over a range of altitudes from 15,000 to 55,000 feet and flight Mach numbers from 0.17 to 0.99.

(a) These data showed that, in general, a change in corrected engine speed, altitude, flight Mach number or exhaust-nozzle area in order to increase the combustor-inlet pressure resulted in an increase in combustion efficiency except at high pressure levels where combustion efficiency was constant. For example, at a flight Mach number of 0.62 and an open exhaust nozzle (area, 534 sq in.) the combustion efficiency decreased from 0.98 to 0.83 as altitude was increased from 15,000 to 55,000 feet at a corrected engine speed of 7600 rpm.

(b) For all flight conditions and exhaust-nozzle areas, combustor total-pressure-loss ratio decreased as the corrected engine speed increased above a corrected engine speed of about 6000 rpm. However, at a constant corrected engine speed, decreasing altitude, or increasing flight Mach number or exhaust-nozzle area, in general, resulted in an increasing total-pressure-loss ratio. At a corrected engine speed of 7000 rpm, an altitude of 35,000 feet, and a flight Mach number of 0.62, an increase in the exhaust-nozzle area from 367 to 534 square inches resulted in an increase of combustor total-pressure-loss ratio from 0.024 to 0.036.

3. A good correlation was obtained when combustion efficiency was presented as a function of combustion parameter P_4T_4/V_b and engine fuel-air ratio. These data indicated that at values of combustion

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parameter below 34,000 pounds-°R-second per cubic foot there was a fuel-air ratio that resulted in an optimum combustion efficiency for a given value of combustion parameter.

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APPENDIX A

SYMBOLS

The following symbols are used in this report:

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A	cross-sectional area, sq ft
c_p	specific heat at constant pressure, Btu/(lb)(°F)
c_v	specific heat at constant volume, Btu/(lb)(°F)
f/a	fuel-air ratio
g	acceleration due to gravity, 32.2 ft/sec^2
H	enthalpy
M	Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft abs
p	static pressure, lb/sq ft abs
q	theoretical dynamic pressure, lb/sq ft abs
R	gas constant, 53.4 ft-lb/(lb)(°R)
T	total temperature, °R
t	static temperature, °R
V	velocity, ft/sec
W_a	air flow, lb/sec
W_f	fuel flow, lb/hr
W_g	gas flow, lb/sec
γ	ratio of specific heats, c_p/c_v
δ	pressure correction factor, P/2116 (total pressure divided by NACA standard sea-level pressure)

η efficiency

θ temperature correction factor, $\gamma T/(1.4)(519)$ (product of γ and total temperature divided by product of γ at standard sea-level temperature and standard NACA sea-level temperature)

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ρ density, $(lb)(sec^2)/ft^4$

Subscripts:

0 free-stream conditions

1 cowl inlet

3 compressor inlet

4 combustor inlet, compressor outlet

5 combustor outlet, turbine inlet

6 turbine outlet

7 exhaust-nozzle outlet

b burner

c compressor

i indicated

t turbine

APPENDIX B

METHODS OF CALCULATION

Air flow. - Air flow was calculated at station 1 (fig. 2) by use of the following equation 2626

$$W_{a,1} = p_1 A_1 \sqrt{\frac{2\gamma_1 g}{(\gamma_1 - 1)Rt_1}\left[\left(\frac{P_1}{p_1}\right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1\right]}$$

Gas flow downstream of the combustor is

$$W_g = W_{a,1} + \frac{W_f}{3600}$$

Combustor dynamic pressure. - In order to calculate a combustor dynamic pressure, based on a combustor maximum cross-sectional area of 6.40 square feet, a combustor Mach number was first calculated with the equation

$$\frac{M_b}{\left(1 + \frac{\gamma_4 - 1}{2} M_b^2\right)^{\frac{\gamma_4 + 1}{2(\gamma_4 - 1)}}} = \frac{W_{a,4}\sqrt{T_4}}{0.776\ A_b P_4 \sqrt{\gamma_4}}$$

then

$$q_b = \frac{\gamma_4 p_4 M_b^2}{2}$$

and

$$p_4 = \frac{P_4}{\left(1 + \frac{\gamma_4 - 1}{2} M_b^2\right)^{\frac{\gamma_4}{\gamma_4 - 1}}}$$

therefore

$$q_b = \frac{\gamma_4 P_4 M_b^2}{2\left(1 + \frac{\gamma_4 - 1}{2} M_b^2\right)^{\frac{\gamma_4}{\gamma_4 - 1}}}$$

[redacted]

Combustor-inlet velocity. - With the use of combustor Mach number M_b, combustor-inlet velocity was determined from the following equation:

$$V_b = M_b\sqrt{\gamma_4 gRt_4}$$

where

$$t_4 = \frac{T_4}{\left(1 + \frac{\gamma_4 - 1}{2} M_b^2\right)}$$

Turbine-inlet temperature. - Turbine-inlet temperature was calculated from the following equation, which assumes compressor and turbine work equal:

$$T_5 = \frac{W_{a,1}\ c_{p,c}}{W_{g,5}\ c_{p,t}} (T_4 - T_1) + T_7$$

Combustion efficiency. - With the assumption that the compressor and turbine work are equal, combustion efficiency is defined as the ratio of the actual enthalpy rise of the gas while passing through the engine to the theoretical increase in enthalpy that would result from complete combustion of the fuel change.

$$\eta_b = \frac{\text{actual enthalpy rise of the gas across the engine}}{\text{heat input}}$$

$$= \frac{3600\left[W_{a,1}\ H_a\right]_{T_1}^{T_7} + \left[W_f\ H_f\right]_{T_b}^{T_7}}{18{,}700\ W_f}$$

where 18,700 Btu per pound of fuel is the lower heating value of the fuel.

Combustor total-density ratio. - From the gas law the total density is

$$\rho = \frac{P}{gRT}$$

then

$$\frac{\rho_4}{\rho_5} = \frac{P_4}{P_5}\frac{T_5}{T_4}$$

REFERENCES

1. Finger, Harold B., Essig, Robert H., and Conrad, E. William: Effect of Rotor- and Stator-Blade Modifications on Surge Performance of an 11-Stage Axial-Flow Compressor. I - Original Production Compressor of XJ40-WE-6 Engine. NACA E52G03, 1952

2. Conrad, E. William, Finger, Harold B., and Essig, Robert H.: Effect of Rotor- and Stator-Blade Modification on Surge Performance of an 11-Stage Axial-Flow Compressor. II - Redesigned Compressor for XJ40-WE-6 Engine. NACA RM E52I10, 1953.

3. Childs, J. Howard: Preliminary Correlation of Efficiency of Aircraft Gas-Turbine Combustors for Different Operating Conditions. NACA RM E50F15, 1950.

4. Pinkel, I. Irving, and Shames, Harold: Analysis of Jet-Propulsion-Engine Combustion-Chamber Pressure Losses. NACA Rep. 880, 1947. (Supersedes NACA TN 1180.)

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TABLE I. - COMBUSTOR PERFORMANCE DATA FOR PROTOTYPE

Run	Altitude (ft)	Ram-pressure ratio P_1/p_0	Flight mach number M_0	Free-stream static pressure p_0 $\left(\frac{\text{lb}}{\text{sq ft abs}}\right)$	Engine speed N (rpm)	Corrected engine speed $N/\sqrt{\theta}$ (rpm)	Compressor-inlet total temperature T_3 (°R)	Combustor-inlet total temperature T_4 (°R)	Combustor-inlet total pressure P_4 $\left(\frac{\text{lb}}{\text{sq ft abs}}\right)$	Calculated combustor-outlet total temperature T_5 (°R)
1	15,000	1.017	0.155	1186	7260	7275	517	857	5573	1630
2		1.022	.176	1184	7260	7205	527	883	5845	1852
3		1.021	.175	1183	6534	6939	460	744	5235	1367
4		1.020	.169	1185	6534	6900	465	754	5357	1467
5		1.022	.176	1185	6534	6913	464	761	5492	1577
6		1.019	.164	1182	6534	6913	464	767	5580	1660
7		1.019	.164	1188	6534	6939	460	772	5791	1767
8		1.297	.621	1181	7260	7376	503	834	7020	1520
9		1.296	.621	1183	7260	7391	501	826	7115	1530
10		1.298	.622	1183	7260	7391	501	840	7295	1610
11		1.292	.616	1183	7260	7398	500	839	7369	1670
12		1.294	.619	1186	7260	7383	502	838	7339	1697
13		1.291	.616	1188	7260	7427	496	843	7546	1737
14		1.292	.616	1186	7260	7369	504	858	7459	1840
15		1.302	.626	1183	7260	7398	500	856	7687	1850
16		1.296	.621	1179	7079	7199	502	816	6833	1460
17		1.294	.619	1183	7079	7190	500	818	6773	1463
18		1.292	.616	1186	7079	7214	500	813	7019	1548
19		1.289	.614	1186	7079	7214	500	833	7171	1667
20		1.291	.616	1184	7079	7192	503	844	7390	1777
21		1.295	.619	1183	6897	6994	505	812	6496	1420
22		1.296	.621	1188	6897	7000	504	815	6736	1500
23		1.289	.614	1191	6897	7063	495	821	6786	1607
24		1.292	.616	1188	6897	7007	503	829	7069	1713
25		1.289	.622	1183	6716	6790	508	806	6106	1395
26		1.295	.619	1183	6716	6803	506	799	6160	1375
27		1.295	.619	1187	6716	6810	505	803	6388	1455
28		1.295	.619	1183	6716	6823	503	801	6510	1473
29		1.288	.613	1194	6716	6803	506	807	6486	1540
30		-----	-----	1188	6716	6817	504	816	6762	----
31		1.292	.617	1188	6534	6599	509	792	5820	1333
32		1.295	.619	1183	6534	6607	508	788	----	1323
33		1.300	.624	1187	6534	6632	504	789	6089	1403
34		1.291	.616	1188	6534	6645	502	788	6187	1476
35		1.298	.622	1183	6534	6619	506	795	6277	1500
36		1.295	.619	1185	6534	6639	503	798	6310	1580
37		1.298	.622	1187	6534	6658	500	811	6720	1803
38		1.298	.622	1182	6171	6233	509	784	5242	----
39		1.301	.625	1188	6171	6257	505	761	5321	1298
40		1.300	.624	1182	6171	6276	502	763	5367	1367
41		1.295	.619	1179	6171	6270	503	768	5537	1450
42		1.296	.621	1183	6171	6288	500	781	5793	1655
43		1.299	.623	1188	5808	5889	505	734	4493	----
44		1.296	.621	1182	5808	5860	510	738	4540	1157
45		1.294	.619	1184	5808	5884	506	736	4634	1230
46		1.295	.619	1187	5808	5854	511	744	4681	1297
47		1.291	.616	1185	5808	5889	505	740	4843	1350
48		1.298	.622	1186	5808	5918	500	752	5083	1527
49		1.291	.616	1190	5082	5112	513	694	3394	1055
50		1.297	.621	1181	5082	5123	511	689	3466	1023
51		1.295	.619	1186	5082	5133	509	687	3485	1082
52		1.301	.625	1187	5082	5143	507	686	3527	1115
53		1.295	.619	1184	5082	5153	505	685	3633	1170
54		1.289	.614	1187	5082	5138	508	699	3636	1233
55		1.288	.613	1190	5082	5179	500	692	3749	1290
56		1.284	.609	1190	3993	4041	507	614	2317	900
57		1.287	.612	1189	3086	3135	503	557	1769	730
58	30,000	1.301	0.625	611	7260	7739	457	775	3864	----
59		1.302	.626	612	7260	7710	460	781	3914	1515
60		1.301	.625	612	7260	7725	458	785	3983	1570
61		1.303	.627	612	7260	7703	461	786	3955	1570
62		1.303	.627	608	7260	7710	460	792	4076	1650
63		1.292	.616	610	7260	7696	462	798	4122	1707
64		1.296	.621	612	7260	7717	459	810	4207	1808
65		1.303	.627	610	7260	7717	459	811	4223	1808
66		1.298	.621	620	7260	7696	462	803	4234	----
67		1.293	.618	612	7260	7739	457	816	4251	1862
68		1.290	.614	612	7260	7725	458	817	4243	1860
69		1.295	.619	611	7260	7739	457	805	4308	1817
70	35,000	1.018	0.160	478	7260	7863	443	772	2477	1553
71		1.014	.141	476	7260	7848	444	790	2573	1715
72		1.012	.130	476	7260	7884	440	800	2647	1818
73		1.013	.136	477	7260	7877	441	802	2650	1825
74		1.017	.155	477	7260	7841	445	803	2678	1877
75		1.020	.169	478	7260	7848	444	802	2702	1863
76		1.018	.160	478	7079	7874	442	759	2429	1499
77		1.019	.168	478	6716	7294	440	735	2294	1420
78		1.019	.168	478	6534	7102	439	722	2216	1371
79		1.021	.173	479	5808	6313	439	676	1776	1249
80		1.022	.176	477	5082	5519	440	635	1353	1198
81		1.022	.176	479	3993	4336	440	571	908	1210
82		1.021	.173	479	3630	3942	440	547	820	1200
83		1.025	.188	479	3086	3351	440	519	705	1200
84		1.293	.618	479	7260	7884	440	767	3087	1493
85		1.288	.613	477	7260	7754	455	784	3049	1533
86		1.292	.616	482	7260	7877	441	769	3160	1550

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J40-WE-8 TURBOJET ENGINE (COMPRESSOR 1, COMBUSTOR A)

Combustor-outlet total pressure P_5 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Fuel flow W_f (lb/hr)	Turbine-outlet total temperature T_6 (°R)	Projected exhaust-nozzle area A_7 (sq in.)	Engine-inlet air flow $W_{a,1}$ (lb/sec)	Engine fuel-air ratio f/a	Combustor total-pressure-loss ratio $\frac{(P_4-P_5)}{P_4}$	Combustor total-pressure-loss coefficient $\frac{(P_4-P_5)}{q_b}$	Combustor total density ratio ρ_4/ρ_5	Combustion efficiency η_b	Combustion parameter $\frac{P_4/T_4}{V_b}$ $\left(\frac{lb\text{-}^{\circ}R\text{-}sec}{ft^3}\right)$	Run
5375	3255	1325	534	80.80	0.0112	0.0355	10.10	1.972	0.990	46,916	1
5674	4135	1535	449	79.66	.0144	.0293	9.096	2.160	.981	52,127	2
5054	2715	1118	536	81.97	.0092	.0348	9.577	1.903	.948	40,491	3
5179	3115	1217	475	80.51	.0107	.0352	9.727	2.013	.945	42,902	4
5323	3520	1316	438	80.18	.0122	.0308	9.548	2.138	.952	45,493	5
5416	3845	1392	414	78.96	.0135	.0294	9.647	2.230	.944	47,820	6
5632	4340	1499	388	80.15	.0150	.0275	9.578	2.354	.957	51,328	7
6759	3760	1224	534	105.44	.0099	.0372	10.00	1.893	.984	56,876	8
6849	3905	1237	511	105.50	.0103	.0374	10.35	1.924	.964	57,958	9
7045	4370	1308	479	106.33	.0114	.0343	9.786	1.985	.967	60,822	10
7119	4615	1366	455	105.51	.0122	.0339	9.921	2.060	.974	62,243	11
7093	4680	1393	449	105.21	.0124	.0335	9.801	2.095	.987	61,959	12
7320	4985	1425	442	106.81	.0130	.0300	9.004	2.124	.988	64,832	13
7428	5530	1527	422	105.12	.0146	.0042	1.216	2.154	.978	63,880	14
7474	5555	1537	416	106.21	.0145	.0277	8.589	2.223	.993	67,529	15
6573	3505	1177	536	104.08	.0094	.0381	9.962	1.860	.966	53,872	16
6521	3495	1182	534	103.46	.0094	.0372	9.730	1.859	.966	53,426	17
6770	4030	1259	479	103.84	.0108	.0355	10.080	1.95	.955	57,905	18
6932	4470	1368	449	103.79	.0120	.0333	9.484	2.070	.991	59,494	19
7158	5150	1474	422	102.58	.0140	.0314	9.748	2.173	.962	64,460	20
6250	3200	1145	534	100.25	.0089	.0379	9.919	-----	.969	51,062	21
6492	3715	1224	479	101.22	.0102	.0362	10.04	1.909	.953	54,538	22
6550	4045	1321	449	100.66	.0112	.0348	9.874	2.028	1.005	55,931	23
6848	4710	1424	422	99.94	.0131	.6315	9.404	2.133	.967	60,234	24
5880	2895	1122	536	95.92	.0084	.0370	9.187	1.794	.981	46,560	25
5928	2885	1107	534	96.80	.0083	.0377	9.587	1.785	.975	47,371	26
6156	3365	1189	479	97.50	.0096	.0363	9.748	1.880	.956	50,627	27
6269	3530	1209	471	98.14	.0100	.0370	10.26	1.910	.957	52,255	28
6254	3700	1270	449	96.43	.0107	.058	10.18	1.979	.970	52,962	29
6540	4345	----	422	------	------	.0328	------	-----	-----	------	30
5589	2605	1077	536	92.80	.0078	.0397	9.830	1.753	.966	44,025	31
----	2625	1068	534	93.31	.0078	------	------	-----	.952	------	32
5884	3035	1145	479	94.58	.0089	.0370	9.657	1.846	.969	47,183	33
5967	3410	1222	449	94.06	.0101	.0356	9.821	1.942	.968	49,246	34
6050	3520	1240	442	94.13	.0104	.0362	10.27	1.958	.959	50,895	35
6103	3860	1321	420	92.70	.0116	.0328	9.583	2.097	.959	51,975	36
6526	4975	1529	367	92.15	.0150	.0289	9.510	2.289	.957	59,193	37
5046	1380	----	534	86.62	.0044	.0374	------	1.690	-----	38,398	38
5125	2395	1060	479	85.96	.0077	.0368	9.159	1.771	.958	39,436	39
5176	2660	1129	449	85.24	.0087	.0356	9.317	1.858	.969	40,909	40
5352	3080	1219	419	84.44	.0101	.0334	9.487	1.953	.958	44,135	41
5618	3910	1408	367	83.62	.0130	.0302	9.358	2.185	.963	48,602	42
4321	1680	941	536	77.66	.0060	.0383	8.776	-----	-----	31,468	43
4365	1710	936	534	77.76	.0061	.0386	8.838	1.631	.939	31,879	44
4313	1945	1005	479	77.58	.0070	.0365	8.802	1.734	.954	33,330	45
4514	2140	1075	449	76.37	.0078	.0357	9.076	1.807	.966	34,757	46
4678	2440	1153	419	76.41	.0089	.0341	9.429	1.888	.952	37,370	47
4920	3095	1302	367	75.89	.0113	.0321	9.645	2.098	.972	41,248	48
3267	1160	874	536	61.25	.0052	.0374	8.194	1.579	.909	22,540	49
3323	1170	843	534	62.73	.0052	.0413	8.984	1.549	.856	22,918	50
3252	1287	908	479	61.83	.0058	.0359	8.224	1.634	.917	23,602	51
3403	1360	934	449	62.26	.0061	.0352	8.267	1.684	.930	24,171	52
3510	1593	990	418	61.90	.0071	.0339	8.425	1.768	.910	25,542	53
3512	1668	1052	392	60.26	.0077	.0341	8.921	1.826	.938	26,522	54
3626	1884	1111	367	60.71	.0086	.0328	9.111	1.927	.950	27,953	55
2257	773	790	536	44.30	.0048	.0259	5.825	1.505	.789	14,648	56
1727	524	682	536	32.70	.0045	.0238	6.176	1.343	.524	11,519	57
3765	2170	1190	536	59.07	0.0102	0.0256	7.071	-----	0.963	30,492	58
3824	2255	1233	505	58.88	.0106	.0230	6.522	1.986	.980	31,449	59
3890	2460	1281	475	59.19	.0115	.0234	6.643	2.048	.972	32,098	60
3852	2430	1284	475	59.87	.0113	.0261	7.410	2.050	.986	31,890	61
3960	2705	1362	451	58.06	.0129	.0285	8.855	2.144	.954	34,452	62
4006	2840	1411	438	56.91	.0139	.0282	9.508	2.201	.939	36,565	63
4112	3130	1492	426	58.69	.0148	.0266	7.197	2.284	.968	36,485	64
4124	3130	1501	426	58.85	.0148	.0235	7.500	2.283	.972	36,681	65
4110	3130	----	426	59.99	.0145	.0293	9.118	-----	-----	36,024	66
4164	3270	1550	418	58.57	.0155	.0205	6.744	2.330	.979	37,570	67
4156	3255	1547	418	58.27	.0155	.0205	6.744	2.325	.972	37,517	68
4193	3235	1515	414	58.02	.0155	.0267	9.350	2.318	.948	38,983	69
2408	1510	1261	534	36.78	0.0114	0.0279	8.118	2.070	0.972	20,072	70
2505	1750	1408	475	36.46	.0133	.0264	8.395	2.230	.992	22,032	71
2585	2010	1503	453	36.29	.0154	.0234	7.949	2.328	.958	23,521	72
2590	2010	1509	453	36.40	.0153	.0227	7.692	2.329	.968	23,573	73
2612	2070	1565	435	36.48	.0158	.0247	8.354	2.396	.990	23,835	74
2632	2071	1558	435	36.84	.0156	.0259	8.750	2.385	.990	24,035	75
2359	1450	1217	534	36.57	.0110	.0288	8.434	2.034	.947	19,528	76
2225	1302	1155	534	35.50	.0102	.0301	8.519	1.992	.932	17,882	77
2148	1229	1122	534	36.66	.0098	.0307	8.718	1.959	.929	17,123	78
1712	935	1024	534	29.08	.0089	.0360	10.00	1.917	.868	13,098	79
1306	789	1011	534	22.40	.0098	.0347	9.792	1.955	.771	9,807	80
885	740	1095	534	14.36	.0143	.0253	9.200	2.178	.614	7,028	81
797	728	1102	534	12.67	.0160	.0281	10.95	2.257	.559	6,416	82
690	683	1130	534	10.33	.0184	.0213	10.00	2.362	.509	5,854	83
3002	1802	1204	534	47.20	.0106	.0275	7.798	-----	.969	24,592	84
2965	1799	1244	517	45.79	.0109	.0276	7.850	2.010	.971	24,557	85
3071	1905	1262	510	47.51	.0111	.0282	8.018	2.074	1.000	25,203	86

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TABLE I. - Continued. COMBUSTOR PERFORMANCE DATA FOR

Run	Altitude (ft)	Ram-pressure ratio P_1/p_0	Flight mach number M_0	Free-stream static pressure p_0 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Engine speed N (rpm)	Corrected engine speed $N/\sqrt{\theta}$ (rpm)	Compressor-inlet total temperature T_3 (°R)	Combustor-inlet total temperature T_4 (°R)	Combustor-inlet total pressure P_4 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Calculated combustor-outlet total temperature T_5 (°R)
87	35,000	1.307	0.631	474	7260	7848	444	774	3185	1592
88		1.301	.625	479	7260	7870	442	784	3315	1680
89		1.305	.629	478	7260	7870	442	784	3311	1700
90		1.298	.622	480	7260	7834	446	787	3273	1698
91		1.303	.627	479	7260	7863	443	794	3413	1833
92		1.299	.623	779	7079	7667	443	758	3054	1460
93		1.295	.619	480	7079	7688	440	754	3023	1450
94		1.294	.619	478	7079	7667	443	758	3098	1530
95		1.275	.600	483	7079	6003	450	768	3065	1534
96		1.286	.611	484	7079	7603	450	782	3179	1668
97		1.297	.621	479	7079	7652	444	784	3330	1780
98		1.293	.618	478	6897	7497	439	743	2929	1400
99		1.292	.616	480	6897	7366	455	756	3893	1403
100		1.304	.628	480	6897	7469	443	747	3036	1470
101		1.299	.623	479	6897	7387	452	775	3071	1628
102		1.297	.621	478	6897	7456	444	770	3229	1710
103		1.297	.621	478	6716	7237	443	734	2869	1360
104		1.297	.621	479	6716	7287	441	730	2847	1355
105		1.300	.624	477	6716	7220	449	742	2890	1428
106		1.303	.627	480	6716	7206	451	761	2956	1550
107		1.295	.619	479	6716	7260	444	756	3096	1655
108		1.291	.616	480	6534	7089	441	722	2771	1330
109		1.293	.618	481	6534	7089	441	717	2730	1300
110		1.329	.651	478	6534	6978	455	734	2830	1330
111		1.282	.607	479	6534	7024	449	731	2754	1385
112		1.301	.625	480	6534	7011	451	740	2843	1492
113		1.292	.616	481	6534	6998	453	748	2797	1500
114		1.300	.624	478	8534	7011	451	745	----	1548
115		1.302	.626	479	6534	7063	444	741	2970	1600
116		1.297	.621	481	6534	7024	449	753	----	1662
117		1.313	.636	478	6534	7018	450	762	3090	1760
118		1.294	.619	481	6171	6696	441	700	2492	1237
119		1.291	.616	482	6171	6634	449	704	2485	1280
120		1.294	.619	480	6171	6604	453	724	2489	1380
121		1.300	.624	479	6171	6665	445	713	2625	1455
122		1.303	.627	479	6171	6665	446	735	2734	1735
123		-----	-----	---	5808	6302	441	672	----	----
124		1.312	.636	476	5808	6296	442	669	----	----
125		1.294	.619	478	5808	6296	442	674	2135	1127
126		1.302	.626	480	5808	6267	446	678	2197	1200
127		1.303	.627	479	5808	6209	454	698	2164	1280
128		1.298	.622	478	5808	6273	445	686	2260	1333
129		1.304	.628	477	5808	6267	446	709	2367	1585
130		1.299	.623	478	5445	5875	446	679	2023	1443
131		1.309	.633	476	5082	5514	441	619	----	----
132		1.298	.622	479	5082	5509	492	625	1583	983
133		1.310	.634	479	5082	7287	447	631	1651	1035
134		1.297	.621	479	5082	5428	455	650	1596	1103
135		1.302	.626	479	5082	5489	445	634	1670	1153
136		1.298	.621	479	5082	5483	446	649	1726	1320
137		1.296	.621	477	3993	4316	444	556	1032	855
138		1.330	.652	477	3086	3333	445	499	763	660
139		1.860	.985	479	7260	7645	468	792	4255	1493
140		1.852	.982	478	7260	7587	475	801	4316	1543
141		1.879	.994	477	7260	7638	469	800	4391	1572
142		1.865	.988	477	7260	7630	470	805	4486	1670
143		1.852	.982	480	7260	7587	475	813	----	1700
144		1.852	.982	481	7260	7360	470	817	4660	1813
145		1.858	.984	478	7260	7623	471	819	4668	1825
146		1.856	.984	480	7260	7667	465	814	----	1825
147		1.843	.978	482	7079	7433	471	781	----	1428
148		1.861	.986	478	7079	7454	468	785	4163	1443
149		1.870	.990	479	7079	7433	471	793	4290	1540
150		1.857	.984	478	7079	7440	470	793	4367	1610
151		1.867	.989	479	7079	7461	467	804	4526	1767
152		1.882	.996	477	6897	7249	470	777	4137	1470
153		1.872	.991	477	6897	7263	468	770	4041	1393
154		1.870	.990	477	6897	7235	472	784	4198	1563
155		1.850	.981	478	8897	7269	467	792	4334	1780
156		1.853	.982	479	6716	7052	471	750	3808	1315
157		1.883	.996	476	6716	7065	469	758	3833	1320
158		1.850	.981	479	6716	7059	470	766	3914	1415
159		1.853	.982	477	6716	7045	472	772	3993	1507
160		1.865	.988	479	6716	7045	472	781	4126	1630
161		1.874	.992	477	6534	6854	472	747	3643	1273
162		1.846	.979	479	6534	6861	471	743	----	1255
163		1.865	.988	477	6534	6880	468	751	3720	1350
164		1.873	.992	479	6534	6854	472	756	3820	1424
165		1.857	.984	481	6534	6808	478	771	3874	1563
166		1.873	.992	476	6534	6887	467	783	4187	1825
167		1.884	.997	477	6171	6473	472	721	3190	1153
168		1.857	.984	478	6171	6480	471	727	3267	1230
169		1.873	.992	478	6171	6480	471	727	3332	1293
170		1.861	.986	478	6171	6480	471	731	3399	1377
171		1.869	.990	479	6171	6517	465	746	3658	1643
172		1.845	.978	478	5808	6104	470	687	2691	1010

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PROTOTYPE J40-WE-8 TURBOJET ENGINE (COMPRESSOR 1, COMBUSTOR A)

Combustor-outlet total pressure P_5 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Fuel flow W_f (lb/hr)	Turbine-outlet total temperature T_6 (°R)	Projected exhaust-nozzle area A_7 (sq in.)	Engine-inlet air flow $W_{a,1}$ (lb/sec)	Engine fuel-air ratio f/a	Combustor total-pressure-loss ratio $\frac{(P_4-P_5)}{P_4}$	Combustor total-pressure-loss coefficient $\frac{(P_4-P_5)}{q_b}$	Combustor total density ratio ρ_4/ρ_5	Combustion efficiency η_b	Combustion parameter $\left(\frac{P_4/T_4}{V_b}\right)$ $\left(\frac{lb,°R,sec}{ft^3}\right)$	Run
3138	2000	1292	478	46.96	0.0118	0.0148	4.434	2.088	0.973	28,211	87
3226	2220	1378	463	47.56	.0130	.0269	8.318	2.202	.984	27,877	88
3227	2220	1400	457	47.49	.0130	.0254	7.850	2.224	1.008	27,843	89
3183	2310	1399	451	46.97	.0137	.0275	8.571	2.219	.953	27,570	90
3320	2605	1530	422	47.24	.0153	.0273	9.208	2.374	.986	30,211	91
2966	1713	1177	536	47.10	.0101	.0288	8.224	1.983	.974	24,187	92
2933	1690	1174	534	46.98	.0100	.0298	8.257	1.982	.984	23,537	93
2928	1882	1248	482	46.45	.0113	.0407	11.89	2.104	.962	24,886	94
3024	1855	1256	479	45.71	.0113	.0134	3.905	2.030	.967	24,780	95
3094	2115	1374	449	46.15	.0127	.0267	8.019	2.192	.992	26,320	96
3165	2450	1483	422	46.65	.0146	.0249	8.218	2.328	.982	28,851	97
2838	1580	1128	534	46.24	.0095	.0311	8.349	----	.961	22,330	98
2800	1563	1141	517	44.81	.0097	.0322	8.857	1.918	.939	22,552	99
2856	1770	1203	484	46.56	.0106	.0267	7.570	2.022	.970	23,854	100
2988	1990	1340	449	44.73	.0124	.0270	8.384	2.159	.974	25,660	101
3164	2275	1428	422	44.94	.0141	.0201	6.842	2.287	.954	28,145	102
2773	1515	1103	536	45.58	.0092	.0335	8.972	1.917	.954	21,739	103
2754	1480	1095	534	45.51	.0090	.0114	8.774	1.919	.970	21,519	104
2906	1620	1172	479	44.63	.0101	----	----	1.914	.958	22,641	105
2875	1825	1280	449	44.32	.0114	.0274	8.020	2.094	.986	23,790	106
3042	2115	1383	422	44.69	.0131	.0175	5.567	2.229	.976	25,934	107
2680	1427	1075	536	44.50	.0089	.0329	8.835	1.905	.952	20,825	108
2625	1360	1046	534	44.16	.0086	.0109	9.408	----	.932	20,362	109
2658	1392	1078	534	44.01	.0088	.0608	17.37	1.929	.939	22,046	110
2788	1512	1138	479	42.88	.0098	----	----	1.872	.934	21,416	111
2750	1623	1190	467	43.56	.0104	.0327	9.300	2.015	.954	22,246	112
2710	1681	1240	449	42.59	.0110	.0311	8.878	2.069	.956	21,990	113
----	1808	1288	435	42.86	.0117	----	----	----	.968	----	114
2925	1935	1331	422	43.68	.0123	.0152	4.688	2.192	.983	24,280	115
----	2085	1395	408	49.11	.0134	----	----	----	.964	----	116
3028	2350	1492	388	43.09	.0151	.0201	7.045	2.357	.957	27,211	117
2597	1165	992	534	41.68	.0078	.0099	9.694	1.841	.932	17,947	118
2326	1245	1047	480	40.14	.0086	.0246	6.778	1.864	.913	18,704	119
2402	1385	1142	449	39.20	.0098	.0350	10.000	1.975	.934	19,260	120
2580	1580	1220	422	40.03	.0110	.0172	5.172	2.077	.947	20,738	121
2677	2070	1476	367	38.51	.0149	.0209	7.308	2.408	.953	23,644	122
----	938	985	536	----	----	----	----	----	.919	----	123
----	954	962	536	38.00	.0070	----	----	----	----	----	124
2051	915	902	534	37.34	.0068	.0092	9.545	1.740	.882	14,693	125
2050	1030	978	481	37.31	.0077	.0273	6.977	1.820	.979	15,576	126
2090	1105	1053	449	35.50	.0086	.0342	9.250	1.899	.919	15,990	127
2199	1250	1122	422	36.04	.0096	.0270	7.722	1.997	.933	17,020	128
2302	1659	1354	367	34.60	.0133	.0275	9.286	2.299	.928	19.799	129
1959	1300	1292	367	30.85	.0117	.0316	10.16	2.194	.918	16,069	130
----	702	861	536	29.98	.0065	----	----	----	----	----	131
1522	680	805	534	29.49	.0064	.0061	8.841	1.636	.742	10,223	132
1628	750	864	481	29.72	.0070	.0139	3.382	1.663	.779	10,990	133
1539	753	916	449	28.13	.0074	.0357	8.906	1.760	.819	10,930	134
1612	855	979	422	28.79	.0082	.0347	9.355	1.884	.856	11,763	135
1676	1032	1133	367	28.00	.0108	.0290	8.475	2.095	.895	12,823	136
996	611	747	536	20.14	.0084	.0349	8.182	1.594	.471	6,369	137
737	670	807	536	15.81	.0118	.0341	8.125	1.370	.172	4,462	138
4127	2330	1205	534	65.51	.0099	.0301	8.101	1.943	1.005	33,499	139
4175	2485	1256	494	64.47	.0107	.0327	9.276	1.991	.990	35,130	140
4265	2615	1278	480	65.69	.0111	.0287	8.182	2.023	.986	35,720	141
4364	2940	1371	449	65.11	.0125	.0272	7.974	2.133	.983	37,114	142
----	2975	1395	442	64.56	.0128	----	----	----	.987	----	143
4543	3425	1507	422	64.88	.0147	.0251	8.014	2.276	.978	40,589	144
4543	3495	1520	414	64.53	.0150	.0268	8.562	2.289	.972	40,700	145
----	3520	1520	414	65.29	.0150	----	----	----	.975	----	146
----	2120	1151	534	64.06	.0092	----	----	----	.988	----	147
4026	2190	1164	534	64.76	.0094	.0329	8.839	1.906	.989	32,596	148
4152	2475	1246	479	64.85	.0106	.0322	8.903	2.007	.990	34,262	149
4246	2740	1323	449	62.90	.0121	.0277	8.345	2.088	.960	36,395	150
4426	3205	1467	422	64.47	.0138	.0221	6.849	2.248	1.004	38,527	151
3994	2255	1194	479	63.58	.0099	.0346	9.597	1.960	.989	32,707	152
3896	2030	1122	534	63.77	.0088	.0359	9.416	1.872	.982	30,930	153
4076	2515	1283	449	75.25	.0093	.0291	5.755	2.004	1.184	28,251	154
4236	2950	1410	422	62.64	.0131	.0249	7.500	2.241	.992	36,185	155
3671	1808	1060	534	61.24	.0082	.0360	9.133	1.818	.954	28,380	156
3692	1815	1055	534	62.03	.0081	.0368	9.097	----	.960	28,373	157
3778	2040	1148	479	61.02	.0093	.0348	9.315	1.913	.977	30,287	158
3863	2300	1239	449	60.86	.0105	.0326	9.286	2.018	.990	31,901	159
4004	2670	1368	422	60.70	.0122	.0296	8.652	2.151	.996	33,626	160
3508	1640	1021	534	59.70	.0076	.0371	9.184	1.777	.958	26,732	161
----	1640	1000	534	59.22	.0077	----	----	----	.910	----	162
3584	1825	1097	479	59.65	.0085	.0366	9.577	1.866	.988	28,131	163
3693	2050	1171	449	59.52	.0096	.0333	9.203	1.949	.972	29,779	164
3733	2395	1301	422	57.99	.0115	.0364	10.37	2.164	.977	30,933	165
4089	3205	1547	367	55.23	.0153	.0234	7.903	2.387	.982	36,816	166
3065	1245	912	534	55.00	.0063	.0392	9.259	1.664	.915	22,417	167
3148	1435	988	479	54.71	.0073	.0364	8.815	1.756	.934	23,349	168
3217	1570	1056	449	54.81	.0080	.0345	8.779	1.843	.967	2,449	169
3299	1855	1147	422	54.38	.0095	.0296	7.937	1.941	.949	25,708	170
3557	2470	1391	367	53.38	.0129	.0276	8.783	2.265	.979	30,439	171
2582	915	800	534	49.51	.0051	.0405	8.651	1.532	.853	17,560	172

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TABLE I. - Concluded. COMBUSTOR PERFORMANCE DATA FOR

Run	Altitude (ft)	Ram pressure ratio P_1/p_0	Flight mach number M_0	Free stream static pressure p_0 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Engine speed N (rpm)	Corrected engine speed $N/\sqrt{\theta}$ (rpm)	Compressor-inlet total temperature T_3 (°R)	Combustor-inlet total temperature T_4 (°R)	Combustor-inlet total pressure P_4 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Calculated combustor-outlet total temperature T_5 (°R)
173	35,000	1.880	0.995	476	5808	6098	471	696	2787	1040
174		1.868	.989	477	5808	6098	471	699	2802	1103
175		1.868	.989	478	5808	6093	472	699	2873	1158
176		1.863	.987	480	5808	6110	469	702	2968	1250
177		1.872	.991	478	5808	6104	470	718	3092	1443
178		1.868	.989	479	5082	5331	472	639	1983	843
179		1.876	.993	478	5082	5336	471	644	2018	855
180		1.854	.983	478	5082	5341	470	645	2043	897
181		1.854	.983	479	5082	5336	471	645	2062	923
182		1.850	.981	481	5082	5341	470	647	2110	987
183		1.863	.987	481	5082	5346	469	656	2247	1130
184	45,000	1.294	0.619	289	7260	7913	437	768	1932	1547
185		1.287	.612	290	7260	7746	456	790	1892	1563
186		1.299	.623	289	7260	7950	433	767	1945	1560
187		1.288	.613	289	7260	7892	439	781	1985	1672
188		1.284	.609	292	7260	7892	439	783	2021	1697
189		1.289	.614	290	7260	7819	447	796	2039	1775
190		1.309	.633	286	7260	7957	432	791	2124	1860
191		1.299	.623	289	7260	7950	433	792	2126	1863
192		1.291	.616	290	7260	7921	436	787	2104	1810
193		1.286	.611	290	7260	7913	437	790	2103	1820
194		1.282	.607	291	7079	7674	442	759	1886	1498
195		1.277	.602	298	7079	7695	439	754	1901	1477
196		1.293	.618	290	7079	7582	452	774	1855	1520
197		1.300	.624	291	7079	7716	437	766	1987	1608
198		1.296	.621	289	7079	7624	447	786	1984	1727
199		1.287	.612	290	7079	7716	437	785	2088	1853
200		1.302	.626	287	6897	7407	450	760	1796	1463
201		1.301	.625	289	6897	7525	436	753	1896	1562
202		1.288	.613	290	6897	7456	444	786	1915	1650
203		1.290	.614	291	6897	7518	437	774	2036	1810
204		1.283	.608	291	6716	7287	441	734	1772	1392
205		1.294	.619	289	6716	7233	447	745	1749	1423
206		1.285	.610	288	6716	7206	451	757	1771	1508
207		1.283	.608	289	6716	7253	445	756	1839	1597
208		1.288	.613	289	6716	7327	436	759	1927	----
209		1.296	.621	289	6534	7096	440	721	1694	1342
210		1.298	.603	290	6534	7057	445	733	1692	1367
211		1.289	.614	291	6534	7050	446	740	1705	1440
212		1.303	.627	290	6534	7089	441	737	1773	1530
213		1.299	.623	287	6534	7129	436	741	1839	1647
214		1.278	.619	289	6171	6671	444	709	1513	1273
215		1.296	.621	291	6171	6689	442	714	1548	1348
216		1.288	.613	289	6171	6689	442	712	1580	1410
217		1.291	.616	289	6171	6733	436	712	1627	1520
218		1.295	.619	288	5990	6541	435	726	1578	1775
219		1.278	.603	291	5808	6302	441	673	1324	1160
220		1.295	.619	291	5808	6290	443	678	1300	1165
221		1.296	.621	291	5808	6307	440	687	1353	1268
222		1.299	.623	287	5808	6296	442	687	1363	1320
223		1.301	.625	288	5808	6342	435	685	1409	1390
224		1.327	.649	286	5808	6337	436	706	1484	1670
225		1.311	.634	288	5445	5935	437	675	1278	1515
226		1.302	.626	289	5082	5524	439	630	976	1075
227		1.300	.624	294	5082	5504	443	634	992	1055
228		1.300	.624	291	5082	5473	447	647	1001	1135
229		1.311	.634	288	5082	5519	440	636	1026	1143
230		1.289	.614	287	5082	5550	435	631	1030	1207
231		1.313	.636	289	5082	5534	438	646	1082	1363
232	50,000	1.284	0.609	224	7260	7928	435	775	1529	1590
233		1.270	.595	227	7260	7942	434	783	1581	1690
234		1.299	.623	223	7260	7913	437	797	1665	1817
235		1.282	.607	225	7260	7950	433	794	1629	1813
236		1.258	.582	229	7260	7950	433	794	1624	1813
237		1.301	.625	222	7260	7928	435	794	1638	1833
238		1.290	.614	222	7260	7928	435	800	1650	1870
239		1.288	.613	225	7260	7950	433	800	1651	1883
240		1.294	.619	222	7260	7928	435	796	1625	1843
241		1.266	.591	234	7079	7752	433	759	1527	1520
242		1.280	.605	225	6718	6991	479	789	1298	1493
243		1.283	.608	226	5808	6011	484	730	937	1250
244		1.300	.624	226	5808	6337	436	681	1051	1197
245		1.295	.619	225	5082	5250	486	678	687	1115
246	55,000	1.302	0.626	163	7260	7863	443	786	1199	1647
247		1.324	.647	161	7260	7870	442	798	1238	1823
248		1.287	.612	174	7260	7928	435	793	1267	1760
249		1.302	.626	175	7260	7928	435	804	1325	1870
250		1.295	.619	178	7260	7942	434	804	1325	1883
251		1.266	.591	188	7260	7863	443	805	1251	1855
252		1.309	.633	168	7260	7865	443	804	1263	1840
253		1.309	.633	168	7260	7841	445	802	1244	1820
254		1.314	.637	162	7079	7681	441	778	1175	1630
255		1.315	.638	164	6716	7314	438	740	1076	1407
256		1.304	.628	169	6534	7116	438	730	1029	1390
257		1.295	.619	168	5808	6325	438	683	790	1207
258		1.279	.604	170	5082	5519	440	634	592	1103

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PROTOTYPE J40-WE-8 TURBOJET ENGINE (COMPRESSOR 1, COMBUSTOR A)

Combustor-outlet total pressure P_5 $\left(\frac{lb}{sq\ ft\ abs}\right)$	Fuel flow W_f (lb/hr)	Turbine-outlet total temperature T_6 (^{o}R)	Projected exhaust-nozzle area A_7 (sq in.)	Engine-inlet air flow $W_{a,1}$ (lb/sec)	Engine fuel-air ratio f/a	Combustor total-pressure-loss ratio $\frac{(P_4-P_5)}{P_4}$	Combustor total-pressure-loss coefficient $\frac{(P_4-P_5)}{q_b}$	Combustor total density ratio ρ_4/ρ_5	Combustion efficiency η_b	Combustion parameter $\left(\frac{P_4/T_4}{V_b}\right)$ $\left(\frac{lb, ^{o}R, sec}{ft^3}\right)$	Run
2656	938	818	534	50.51	0.0052	0.0401	8.538	1.556	0.887	18,185	173
2690	1049	880	479	49.89	.0058	.0400	8.960	1.644	.935	18,905	174
2766	1200	937	449	49.65	.0067	.0373	8.992	1.727	.920	20,139	175
2836	1413	1030	422	49.84	.0079	.0448	11.37	1.865	.946	21,410	176
2994	1850	1227	367	48.14	.0107	.0317	8.991	2.076	.955	23,815	177
1902	521	680	534	40.77	.0035	.0409	7.570	1.375	.785	11,630	178
1932	528	685	534	41.30	.0036	.0426	7.890	1.387	.791	11,879	179
1959	600	723	479	40.61	.0041	.0411	8.155	1.448	.792	12,462	180
1982	680	753	449	40.17	.0047	.0388	8.081	1.489	.784	12,875	181
2037	767	815	422	40.14	.0053	.0346	7.374	1.581	.859	13,358	182
2171	991	951	367	39.82	.0069	.0338	8.172	1.783	.915	15,265	183
1881	1213	1257	536	28.71	0.0117	0.0264	7.727	2.069	0.950	15,621	184
1913	1200	1265	534	27.83	.0120	------	------	1.956	.915	15,507	185
1918	1239	1263	521	28.90	.0119	.0139	4.091	2.063	.942	15,717	186
1936	1374	1368	480	28.52	.0134	.0247	7.656	2.195	.944	16,653	187
1975	1449	1395	467	28.70	.0140	.0228	7.302	2.217	.935	17,240	188
1984	1520	1471	449	28.26	.0149	.0270	8.871	2.292	.946	17,795	189
2076	1687	1549	435	28.88	.0162	.0226	7.869	2.905	.957	19,064	190
2074	1687	1550	435	28.85	.0162	.0245	8.525	2.411	.960	19,093	191
2053	1640	1504	435	28.80	.0158	.0242	8.500	2.357	.939	18,829	192
2056	1649	1513	435	28.72	.0160	.0224	7.833	2.357	.933	18,851	193
1843	1141	1216	536	28.24	.0112	.0228	6.719	2.020	.928	15,159	194
1854	1144	1200	536	28.87	.0110	.0247	7.015	2.009	.932	15,009	195
1802	1130	1234	534	27.86	.0113	.0288	8.154	-----	.934	14,837	196
1914	1310	1315	480	28.62	.0126	.0270	8.154	2.157	.945	16,104	197
1932	1450	1430	451	28.04	.0144	.0262	8.387	2.256	.940	16,952	198
2013	1650	1551	422	28.24	.0162	.0359	12.93	2.449	.957	18,952	199
1746	1057	1193	534	27.59	.0106	.0278	7.937	1.980	.939	14,241	200
1849	1223	1281	480	28.33	.0120	.0248	7.460	2.127	.955	15,394	201
1867	1350	1371	451	27.64	.0136	.0251	8.000	2.203	.933	16,176	202
1985	1570	1515	422	28.13	.0155	.0251	8.793	2.399	.986	18,066	203
1723	1000	1133	536	27.48	.0101	.0277	7.903	1.950	.915	13,804	204
1694	1000	1158	534	26.98	.0103	.0315	9.016	1.979	.928	13,726	205
1721	1101	1239	479	26.45	.0116	.0282	8.197	2.050	.917	14,218	206
1790	1245	1325	451	26.96	.0128	.0267	8.305	2.170	.937	15,178	207
1864	1440	----	422	30.15	.0133	.0156	4.167	-----	-----	14,852	208
1649	928	1092	536	26.61	.0097	.0266	7.500	1.912	.895	13,081	209
1636	930	1109	534	26.67	.0097	.0331	9.180	1.930	.912	12,991	210
1650	1020	1183	479	26.23	.0108	.0323	9.167	2.011	.911	13,341	211
1728	1152	1267	451	26.56	.0120	.0254	7.627	2.130	.933	14,239	212
1806	1319	1381	422	26.55	.0138	.0180	5.893	2.264	.938	15,489	213
1456	789	1031	534	24.51	.0089	.0376	10.18	1.865	.874	11,269	214
1492	887	1103	479	29.65	.0100	.0362	10.00	1.959	.882	11,727	215
1531	950	1174	451	24.42	.0108	.0310	9.074	2.043	.906	12,296	216
1587	1095	1278	422	24.30	.0125	.0246	7.843	2.189	.910	13,229	217
1540	1315	1525	367	21.81	.0167	.0241	9.048	2.505	.907	13,998	218
1260	673	938	536	22.43	.0083	.0483	12.55	1.812	.794	9,474	219
1254	673	937	534	22.19	.0084	.0354	9.020	1.781	.777	9,217	220
1305	773	1038	479	22.08	.0097	.0355	9.796	1.914	.823	10,054	221
1315	794	1100	451	21.72	.0102	.0352	10.000	1.991	.853	10,272	222
1374	895	1171	422	21.92	.0113	.0249	7.447	2.081	.872	10,911	223
1448	1173	1435	367	21.01	.0155	.0243	8.781	2.424	.886	12,783	224
1241	942	1303	367	18.96	.0138	.0290	9.737	2.311	.857	10,425	225
956	593	886	536	17.92	.0092	.0205	4.878	1.742	.645	6,414	226
1026	585	871	534	18.10	.0090	------	------	1.609	.651	6,624	227
968	663	942	479	17.88	.0103	.0330	8.047	1.814	.632	6,751	228
992	627	953	451	17.35	.0100	.0331	9.189	1.857	.679	7,338	229
967	683	1029	422	17.33	.0109	.0612	17.50	2.038	.721	7,442	230
1059	779	1180	367	17.06	.0127	.0302	9.706	2.176	.795	8,456	231
1486	995	1290	536	22.36	0.0124	0.0281	8.431	2.111	0.931	12,590	232
1543	1172	1386	483	22.49	.0137	.0240	7.600	2.211	.949	13,486	233
1621	1232	1502	471	22.33	.0153	.0264	9.565	2.342	.968	15,230	234
1594	1285	1501	455	22.37	.0160	.0215	7.292	2.333	.926	14,419	235
1586	1282	1501	455	22.28	.0160	.0234	7.917	2.338	.926	14,375	236
1596	1282	1522	447	22.33	.0159	.0257	8.936	2.370	.950	14,728	237
1612	1356	1559	443	22.17	.0170	.0230	8.261	2.393	.918	15,127	238
1612	1351	1565	443	22.30	.0168	.0236	8.298	2.411	.940	14,901	239
1603	1292	1532	442	22.28	.0161	.0135	4.583	2.346	.944	14,399	240
1467	953	1234	536	22.76	.0116	.0262	7.692	2.057	.931	12,274	241
1254	801	1214	536	19.45	.0114	.0539	9.565	1.958	.875	10,480	242
896	659	1027	536	15.11	.0121	.0438	11.39	1.790	.602	6,990	243
1014	674	969	536	17.52	.0107	.0352	9.487	1.822	.655	7,672	244
660	568	931	534	11.92	.0132	.0393	9.643	1.712	.449	4,740	245
1164	908	1346	527	16.57	0.0152	0.0292	10.00	2.158	0.813	10,559	246
1207	1002	1517	487	16.70	.0167	.0251	8.857	2.343	.897	11,153	247
1226	1002	1448	473	17.39	.0160	.0328	11.08	2.293	.873	11,209	248
1283	1123	1557	455	17.74	.0170	.0317	11.35	2.402	.891	12,173	249
1295	1125	1568	455	17.66	.0177	.0227	8.108	2.396	.894	12,173	250
1221	1060	1544	455	16.46	.0179	.0240	8.824	2.361	.857	11,685	251
1231	1065	1534	451	16.81	.0176	.0253	9.143	2.349	.860	11,604	252
1220	1012	1509	---	16.85	.0167	.0193	6.667	2.314	.888	11,239	253
1142	854	1337	536	16.53	.0143	.0281	9.167	2.156	.858	10,139	254
1050	779	1138	536	16.12	.0134	.0242	7.222	1.948	.697	8,663	255
987	744	1126	536	15.69	.0132	.0408	12.00	1.985	.700	8,105	256
762	655	977	536	12.85	.0142	.0355	9.655	1.832	.504	5,855	257
571	637	924	538	9.96	.0178	.0355	10.00	1.804	.366	4,288	258

NACA

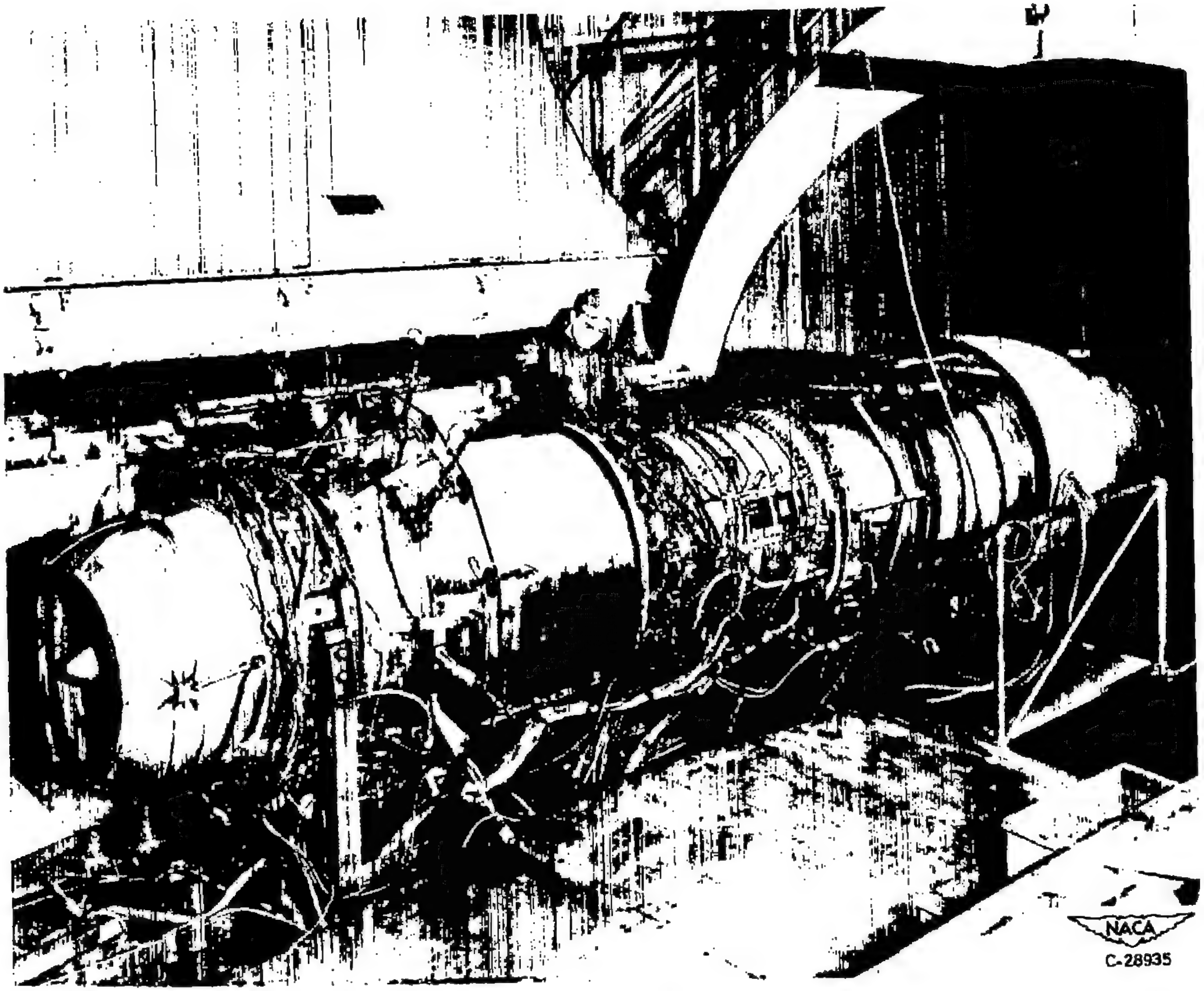
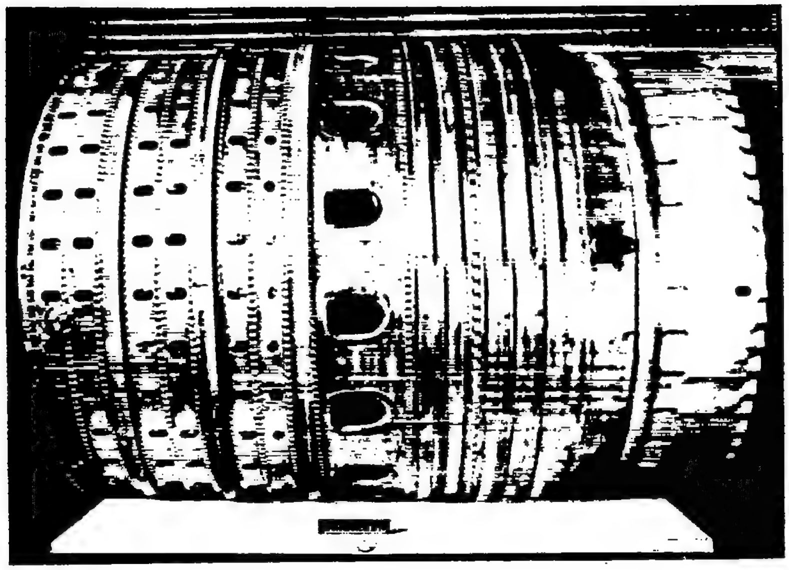


Figure 1 - Engine installation in altitude wind tunnel test section.

Air flow →



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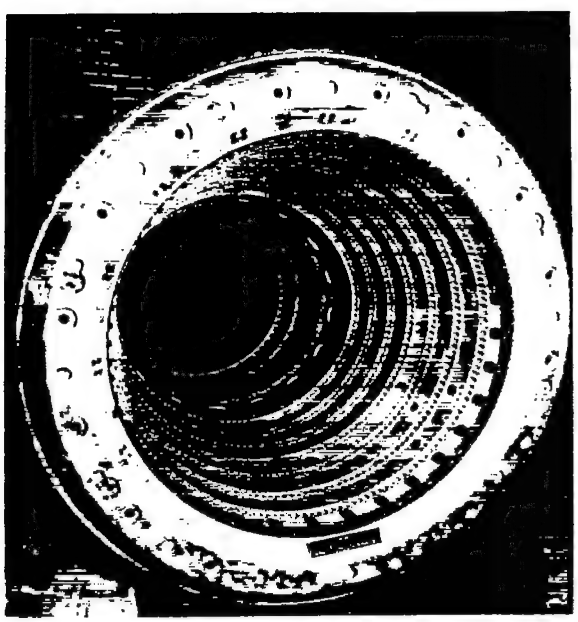
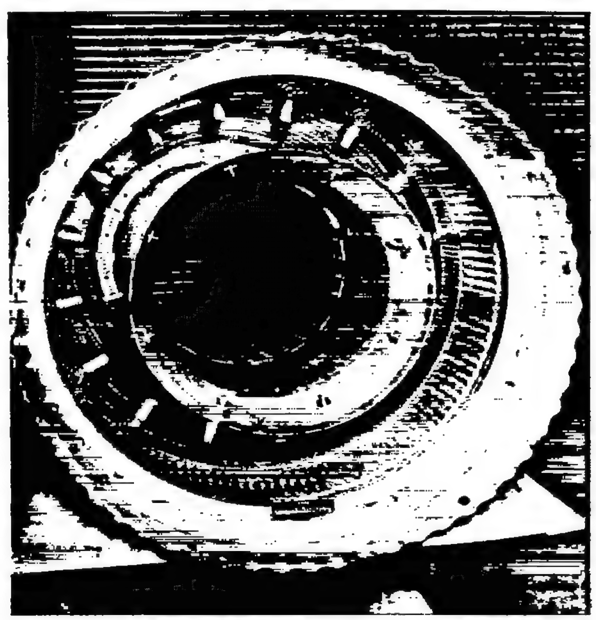
(a) Side view.

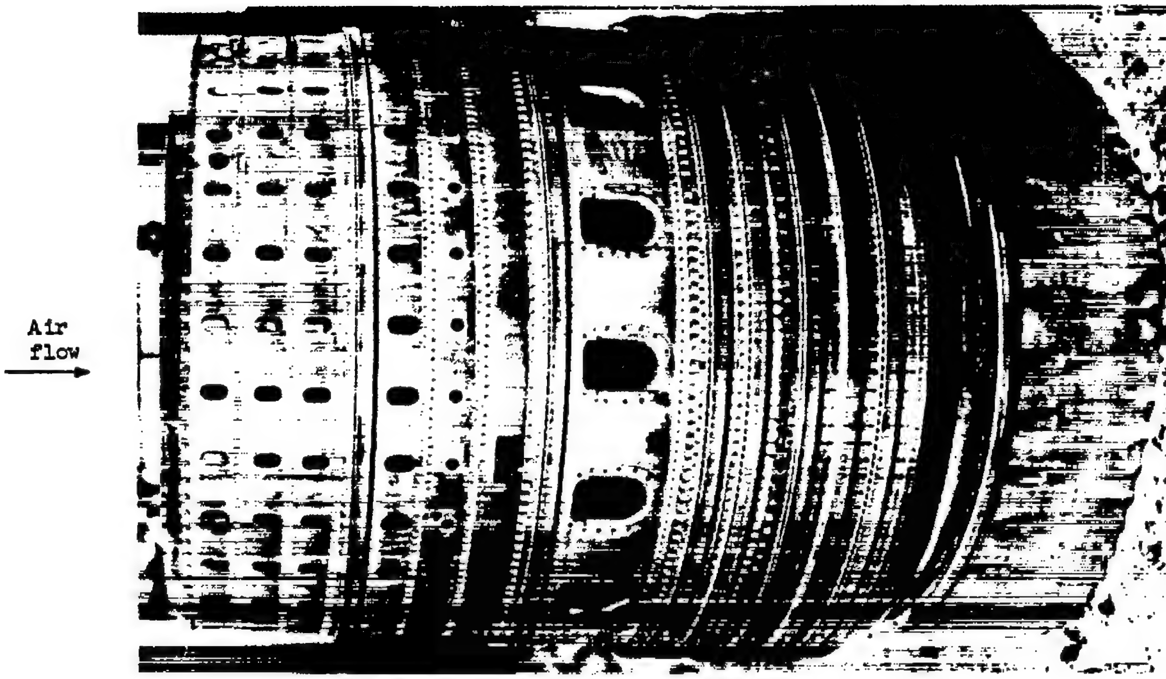


(b) Front view.

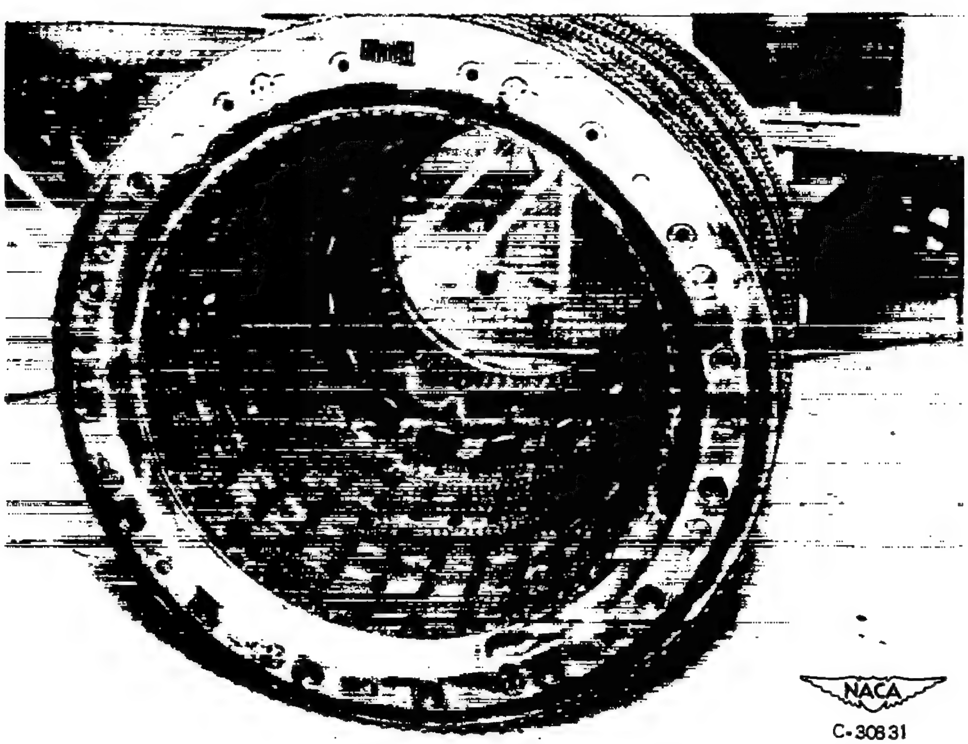


(c) Rear view.

Figure 2. - Engine combustor A.



(a) Side view.



(b) Front view.

Figure 3. - Engine combustor B.

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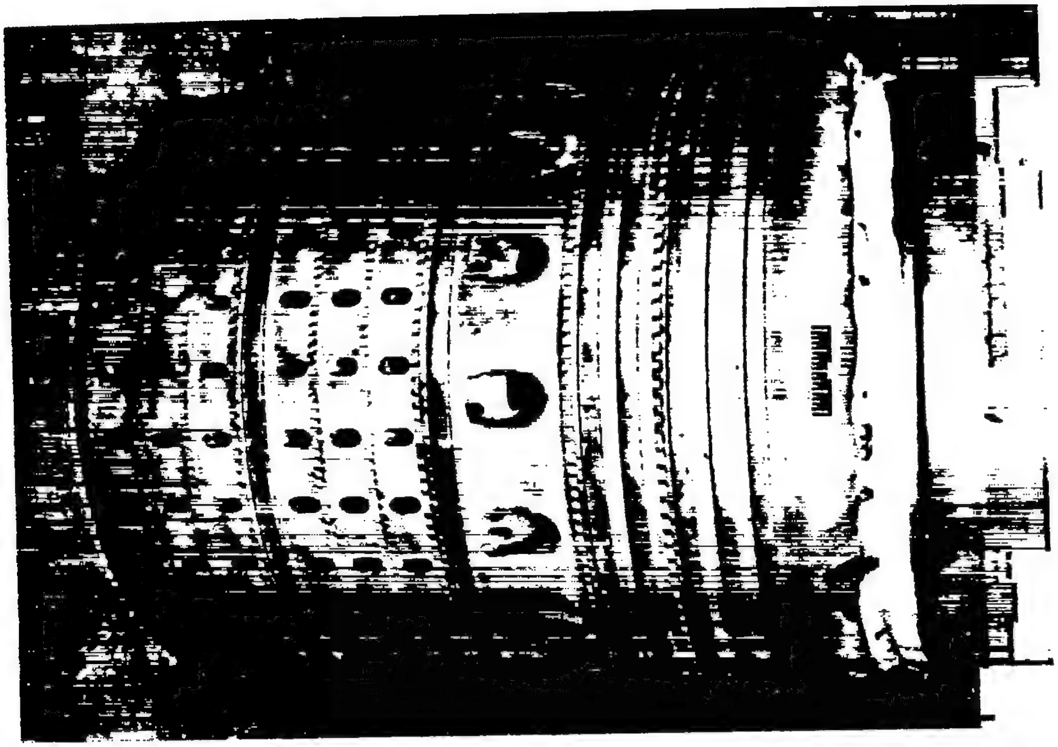
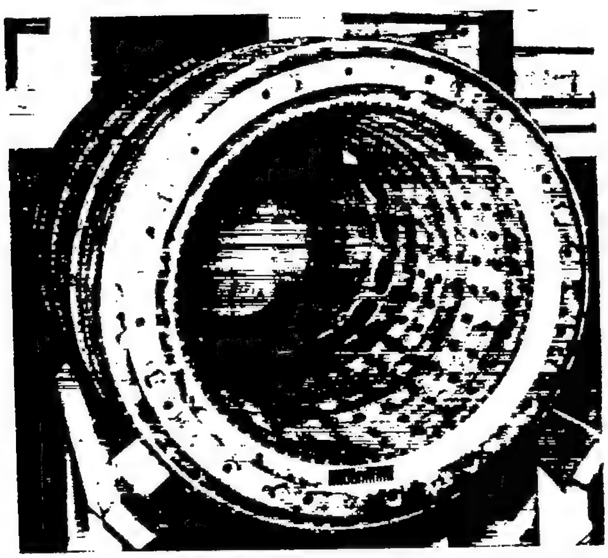
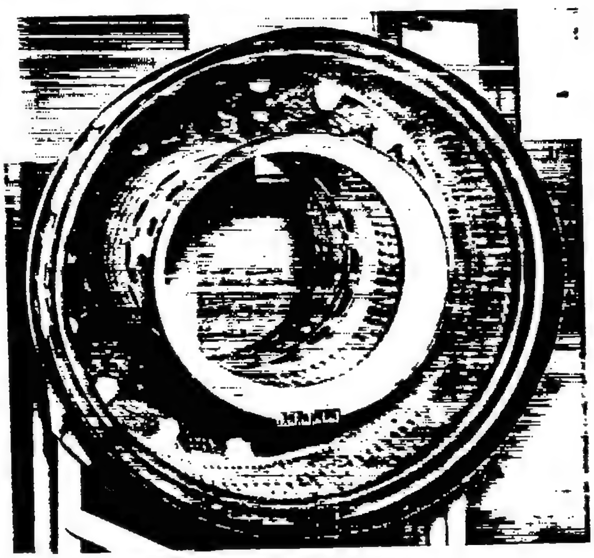
Air flow



(a) Side view.





(b) Front view.

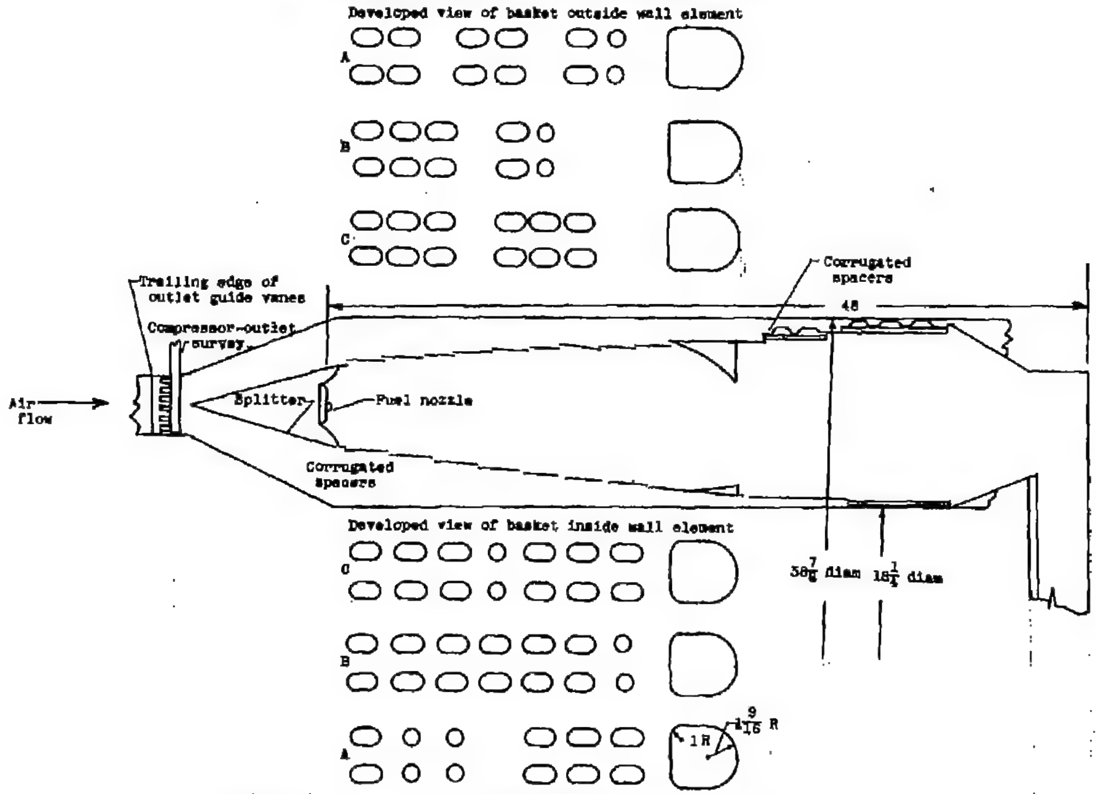
C-30830

(c) Rear view.

Figure 4. - Engine combustor C.

[redacted]

Combustor	Hole openings								
A	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	------	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	------	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}$	$3\frac{1}{8}\times4$
B	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	------	$\frac{7}{8}\times1\frac{1}{2}$	$\frac{5}{8}$	------	-	$3\frac{1}{8}\times4$
C	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	------	$\frac{7}{8}\times1\frac{1}{2}$	$\frac{7}{8}\times1\frac{1}{2}$	$\frac{7}{8}\times1\frac{1}{2}$	-	$3\frac{1}{8}\times4$



Combustor	Hole openings							
C	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$3\frac{1}{8}\times4$
B	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{5}{8}$	$3\frac{1}{8}\times4$
A	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}$	$\frac{3}{4}$	------	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$\frac{3}{4}\times1\frac{5}{16}$	$3\frac{1}{8}\times4$

Figure 5. - Combustor-basket configurations. All dimensions are in inches.

2626

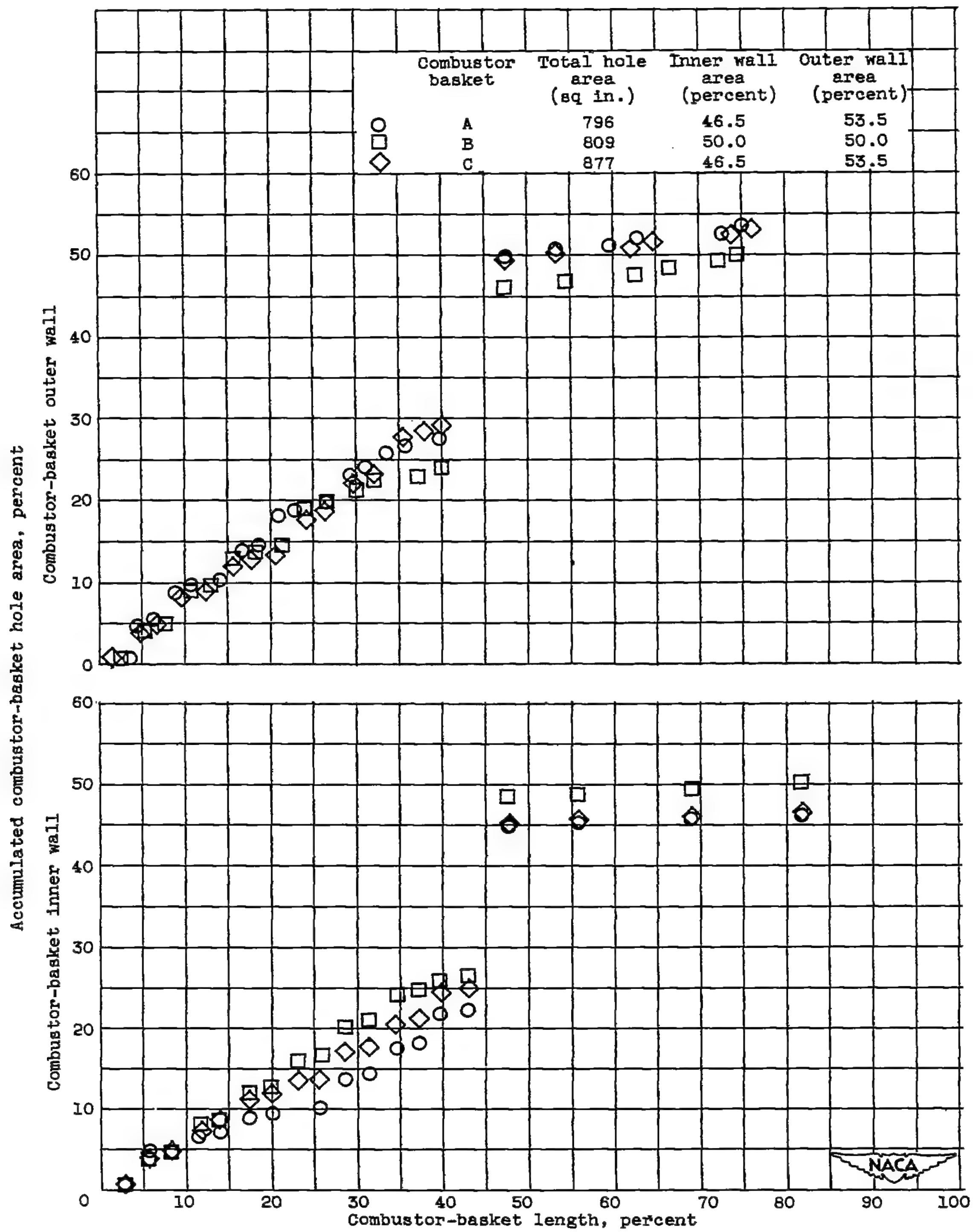
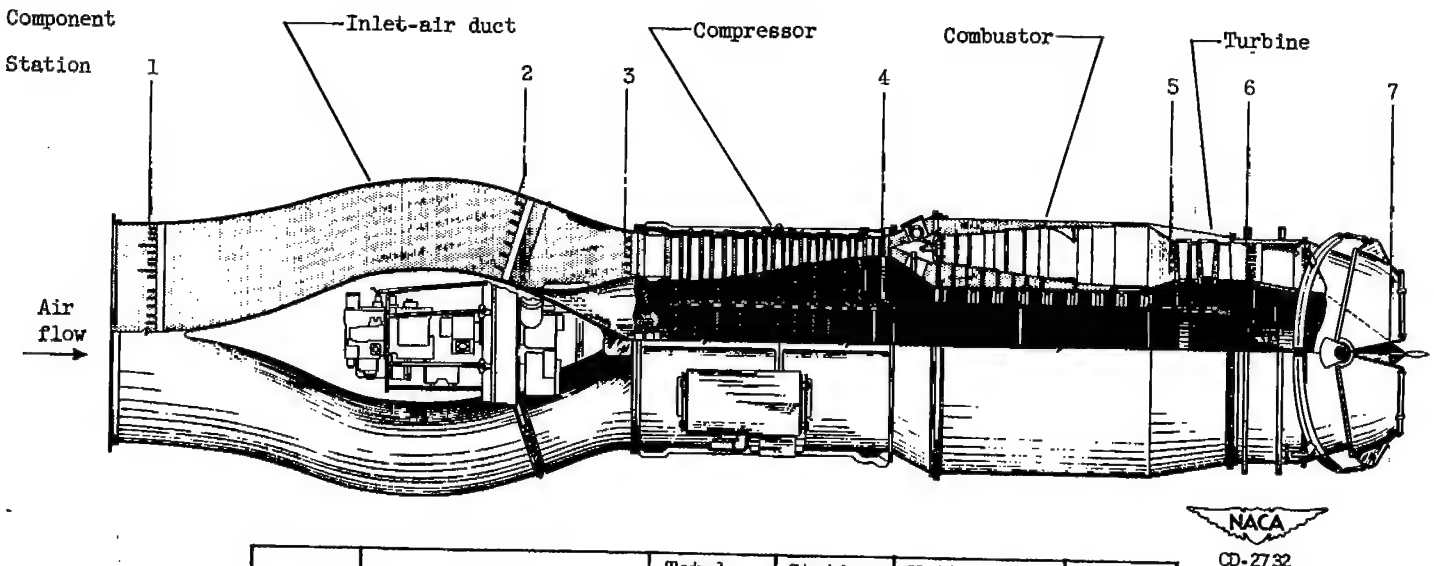


Figure 6. - Percentage of total open area of combustor baskets.



Station	Location	Total-pressure tubes	Static-pressure tubes	Wall static-pressure orifices	Thermo-couples
1	Inlet-air duct	29	12	6	10
2	Engine inlet	18	0	4	0
3	Compressor inlet	23	3	7	0
4	Compressor outlet	18	0	3	6
5	Turbine inlet	5	0	0	10*
6	Turbine outlet	20	0	8	24
7	Exhaust-nozzle outlet	16	2	8	0

*Sonic flow probes

Figure 7. - Top view of turbojet-engine installation showing stations at which instrumentation was installed.

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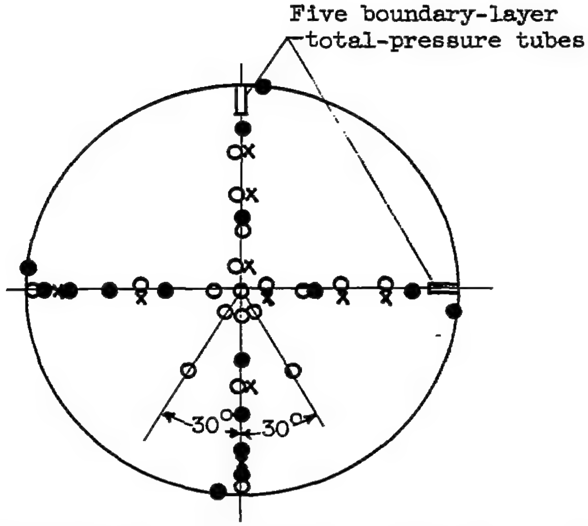
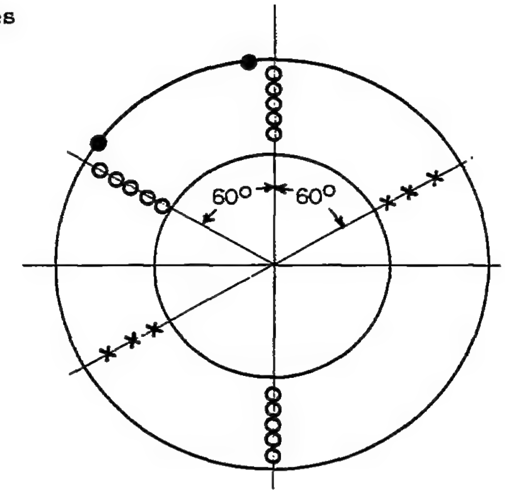
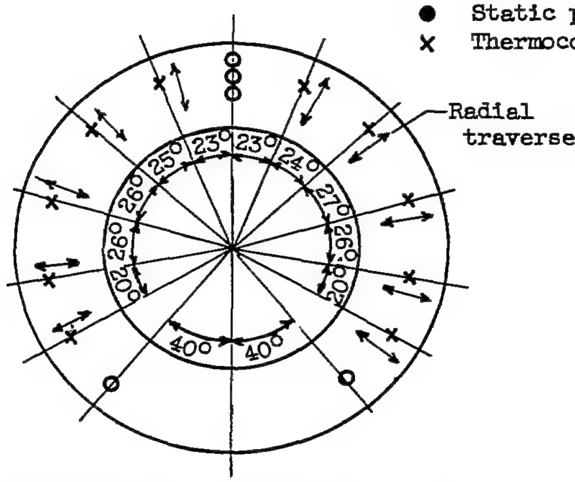


(a) Station 1, cowl inlet. Diameter, 34 inches; location, 6 inches downstream of cowl-inlet flange.

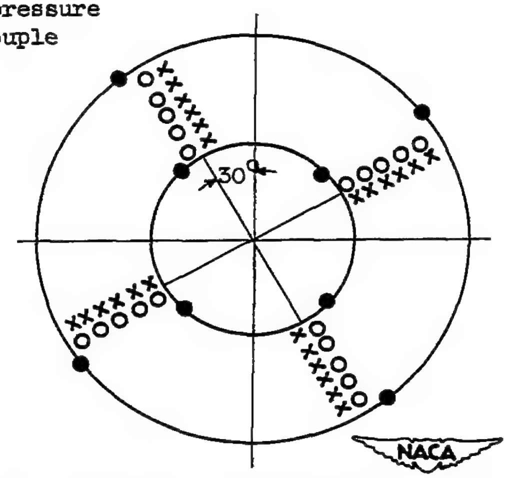


(b) Station 4, compressor outlet. Passage height, $3\frac{1}{8}$ inches; location, $\frac{1}{2}$ inch downstream of trailing edge of fixed vanes.

O Total pressure
● Static pressure
X Thermocouple



(c) Station 5, turbine inlet. Passage height, $6\frac{3}{4}$ inches; location, $1\frac{3}{4}$ inches upstream of leading edge of first stage turbine-nozzle diaphragm.



(d) Station 6, turbine outlet. Passage height, $5\frac{5}{8}$ inches; location, $3\frac{3}{8}$ inches downstream of trailing edge of turbine rotor.

Figure 8. - Location of instrumentation. Viewed looking downstream.

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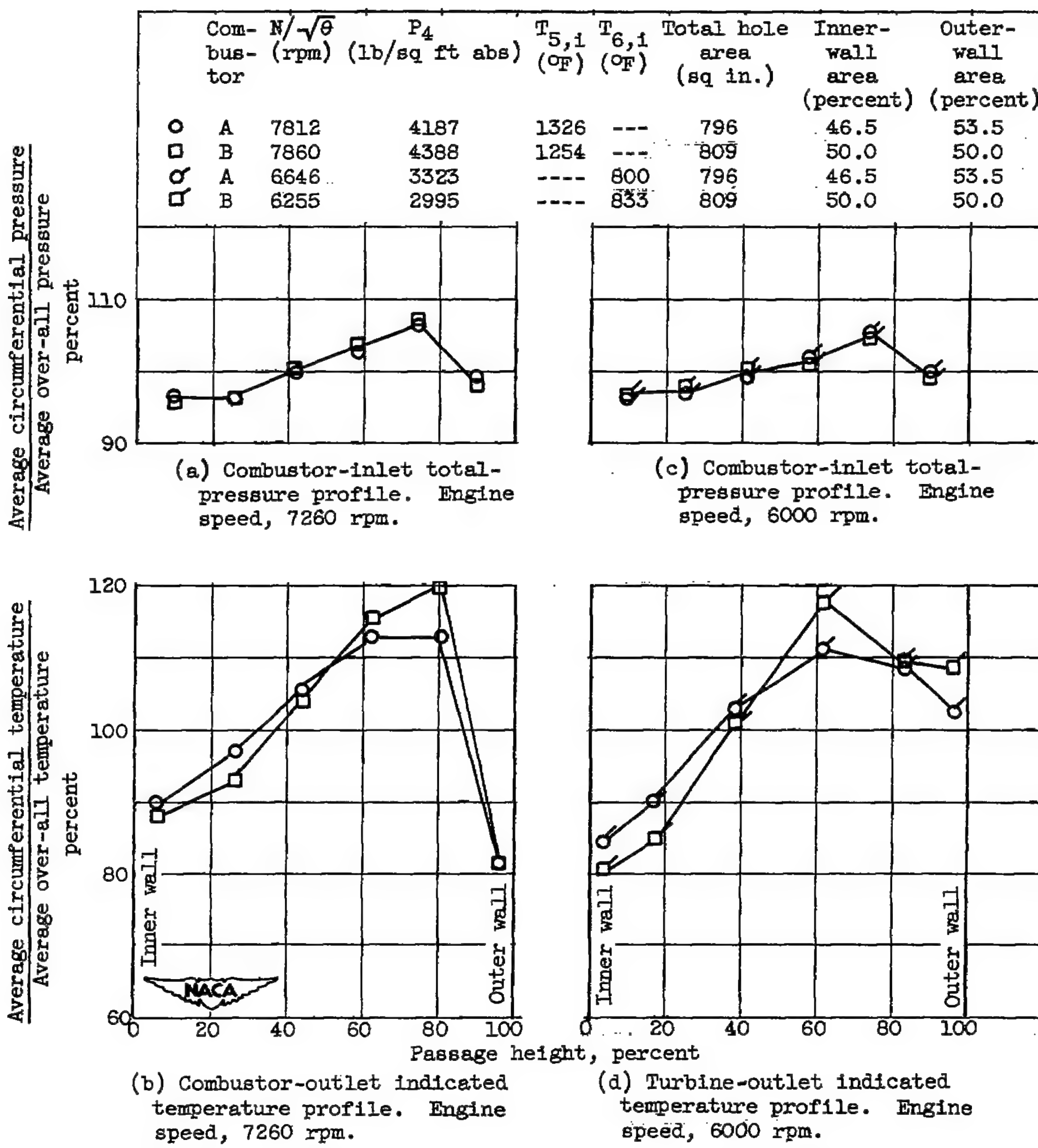
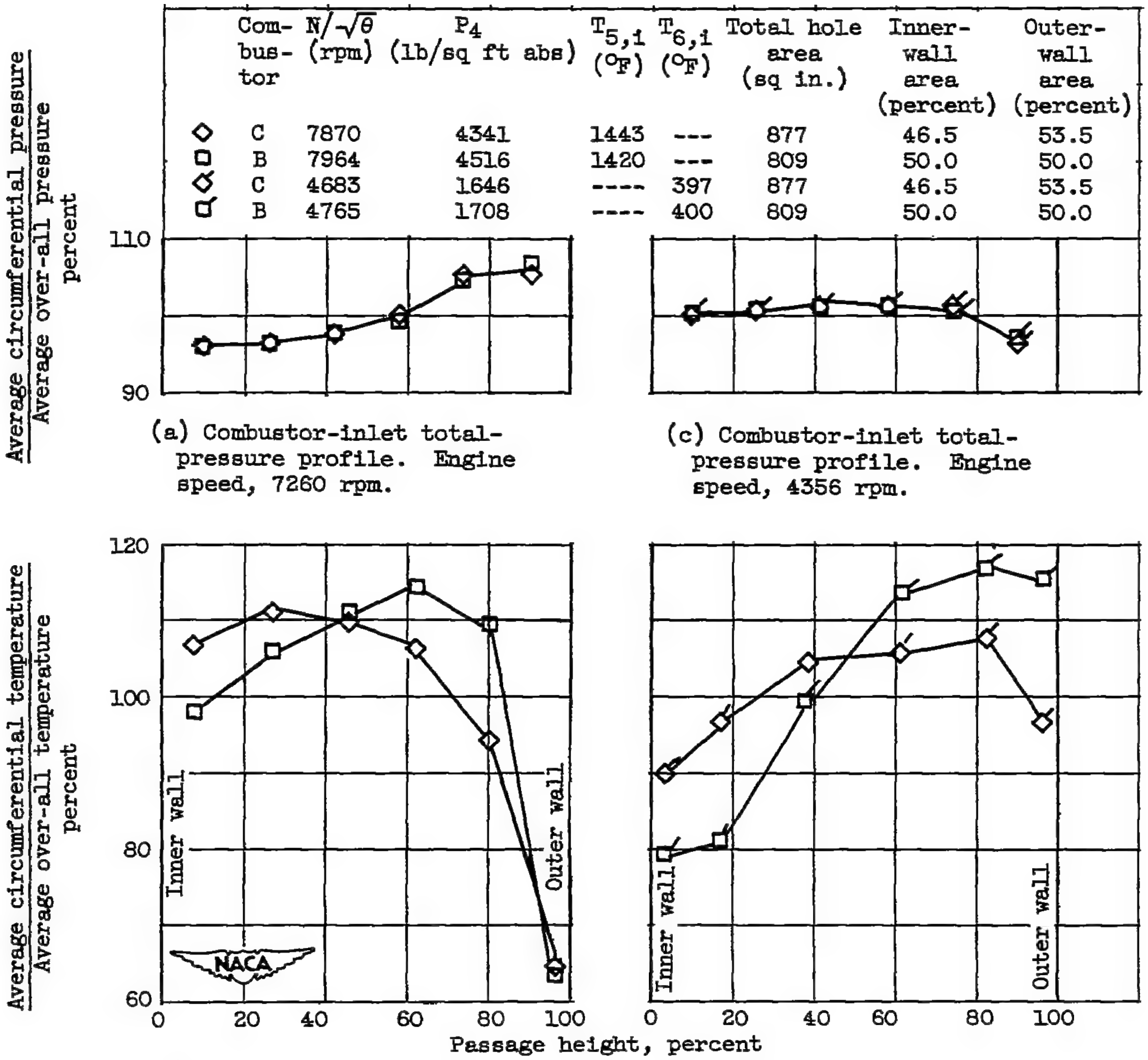


Figure 9. - Effect of combustors on combustor-outlet indicated temperature profiles. Altitude, 30,000 feet; flight Mach number, 0.62; compressor, 1.

2626



(a) Combustor-inlet total-pressure profile. Engine speed, 7260 rpm.

(c) Combustor-inlet total-pressure profile. Engine speed, 4356 rpm.

(b) Combustor-outlet indicated temperature profile. Engine speed, 7260 rpm.

(d) Turbine-outlet indicated temperature profile. Engine speed, 4356 rpm.

Figure 10. - Effect of combustors on combustor-outlet indicated temperature profiles. Altitude, 30,000 feet; flight Mach number, 0.62; compressor, 3.

2626

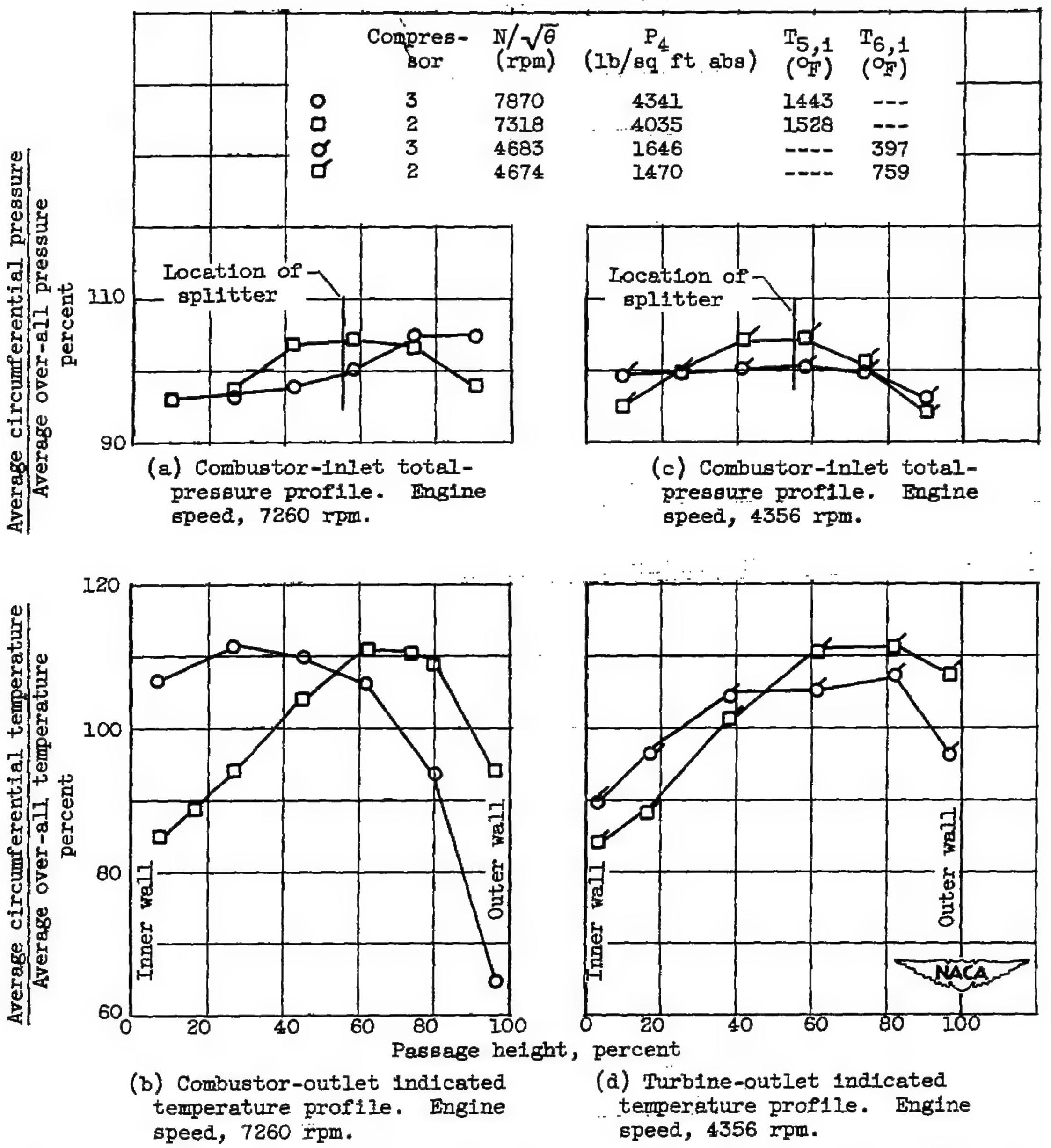
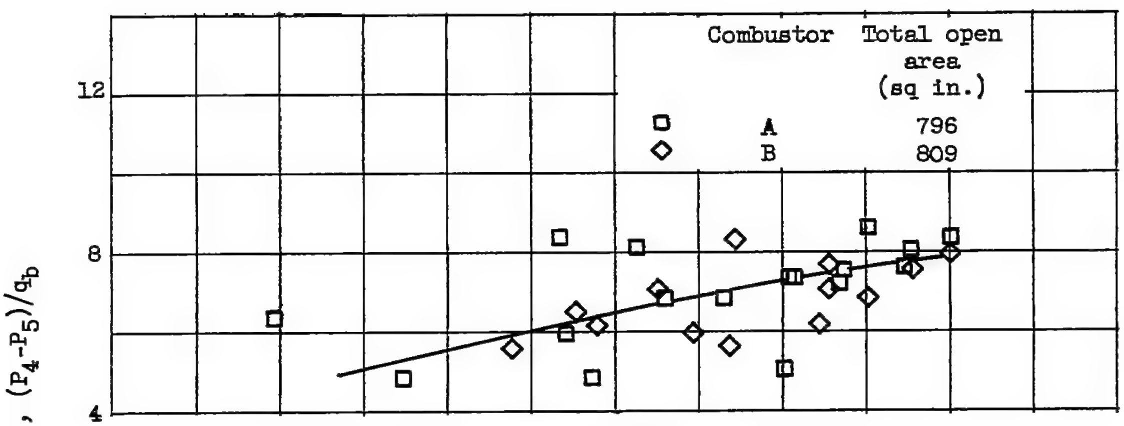
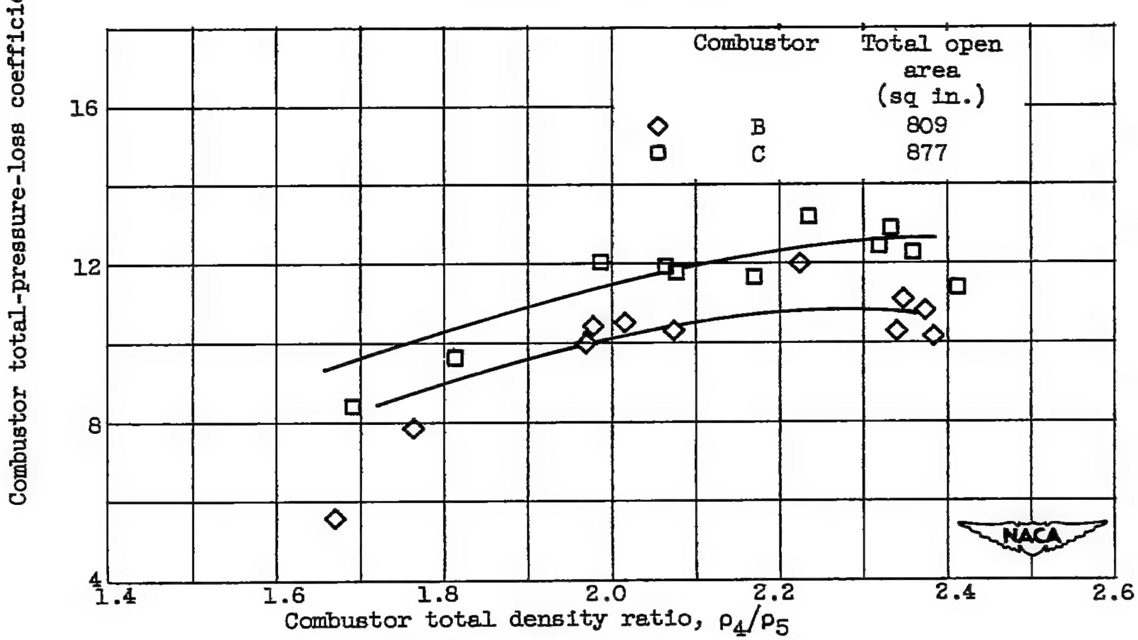


Figure 11. - Effect of combustor inlet-air pressure profiles on combustor-outlet indicated temperature profiles. Altitude, 30,000 feet; flight Mach number, 0.62; combustor, C.

2626



(a) Effect of combustors with compressor 1.



(b) Effect of combustors with compressor 3.

Figure 12. - Variation of combustor total-pressure-loss coefficient with density ratio for several combustors. Altitude, 30,000 feet; flight Mach number, 0.62.

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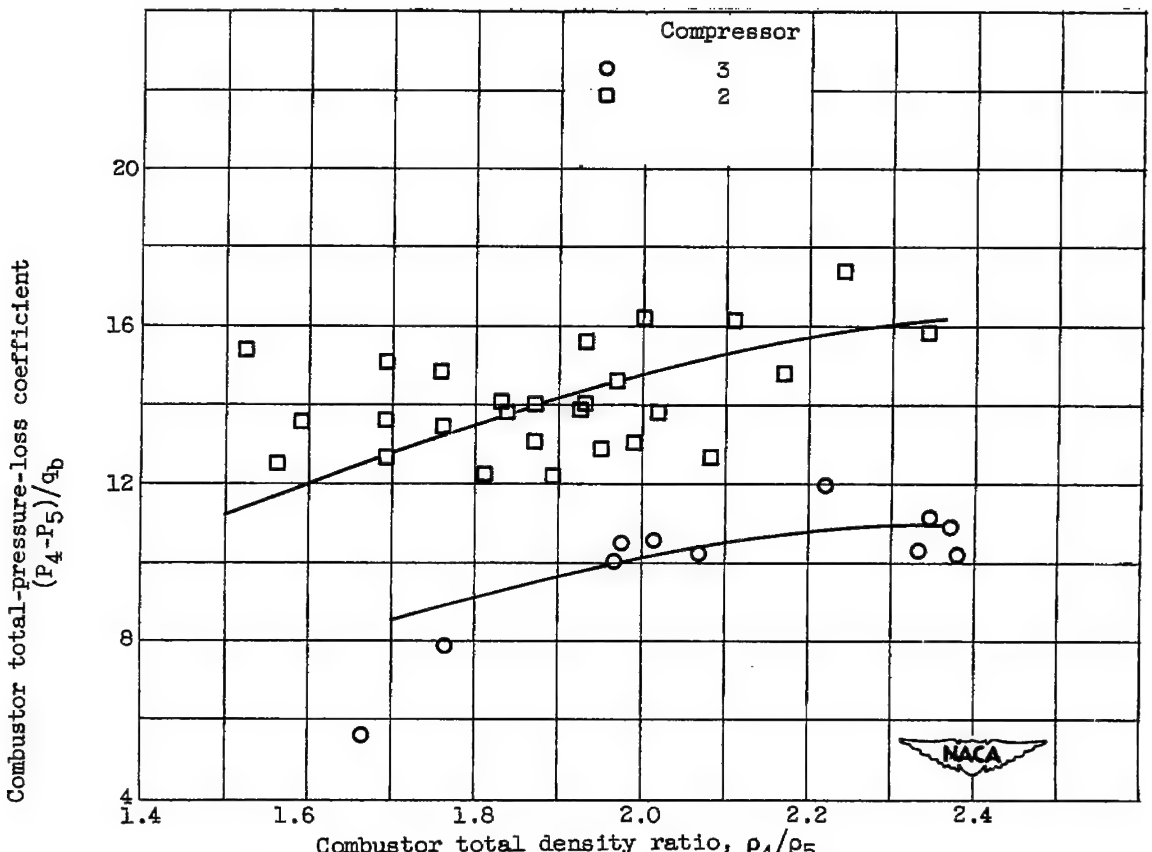

(c) Effect of inlet-air pressure profiles with combustor C.

Figure 12. - Concluded. Variation of combustor total-pressure coefficient with density ratio for several combustors. Altitude, 30,000 feet; flight Mach number, 0.62.

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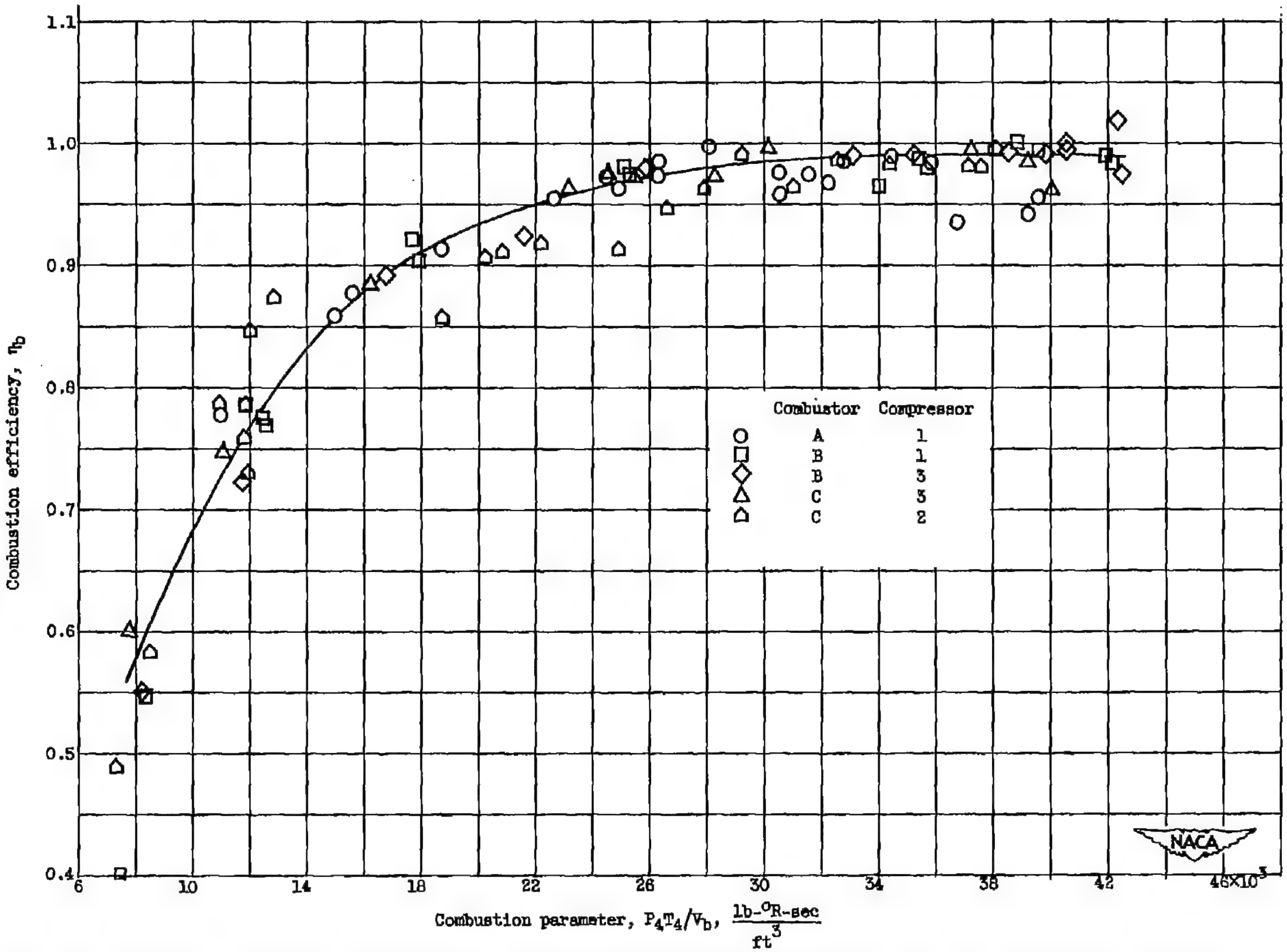

Figure 13. - Variation of combustion efficiency with combustion parameter for three combustors and three compressors. Altitude, 30,000 feet; flight Mach number, 0.62.

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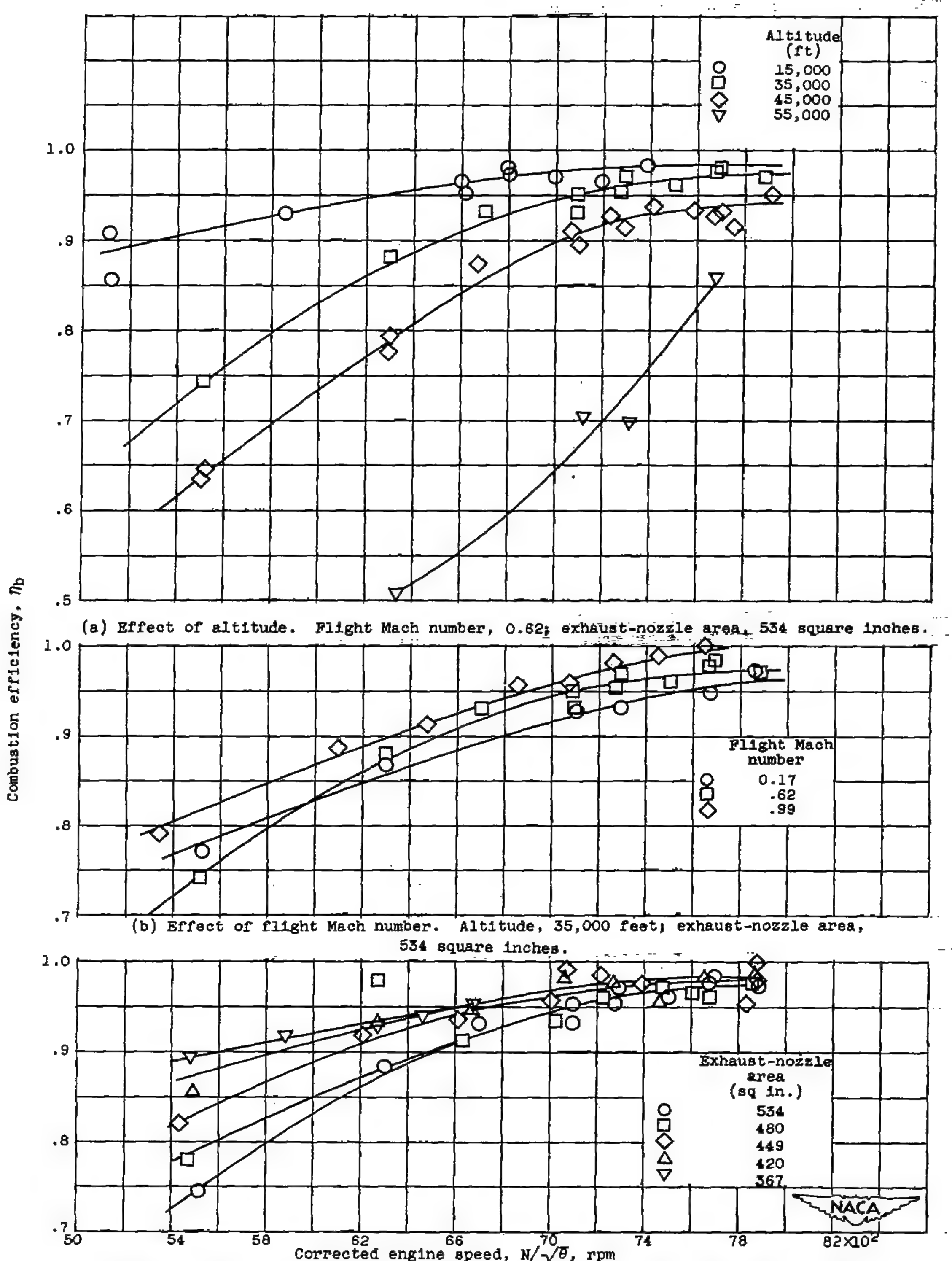

(a) Effect of altitude. Flight Mach number, 0.62; exhaust-nozzle area, 534 square inches.

(b) Effect of flight Mach number. Altitude, 35,000 feet; exhaust-nozzle area, 534 square inches.

(c) Effect of exhaust-nozzle area. Altitude, 35,000 feet; flight Mach number, 0.62.

Figure 14. - Variation of combustion efficiency with corrected engine speed. Prototype J40-WE-8 turbojet engine (compressor 1, combustor A).

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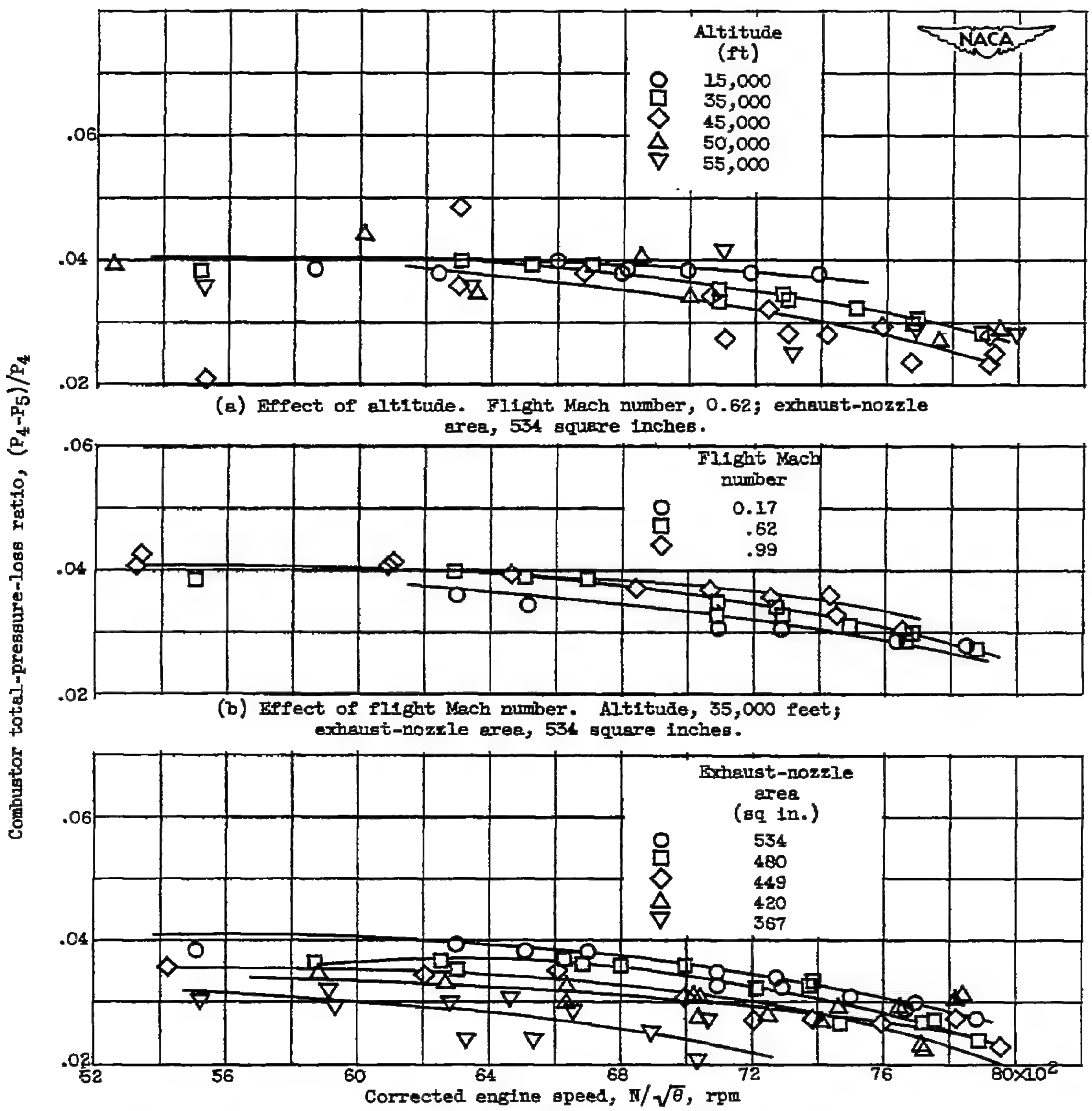
2626



(c) Effect of exhaust-nozzle area. Altitude, 35,000 feet; flight Mach number, 0.62.

Figure 15. - Combustor pressure-loss characteristics in terms of engine parameters. Prototype J40-WE-8 turbojet engine (compressor 1, combustor A).

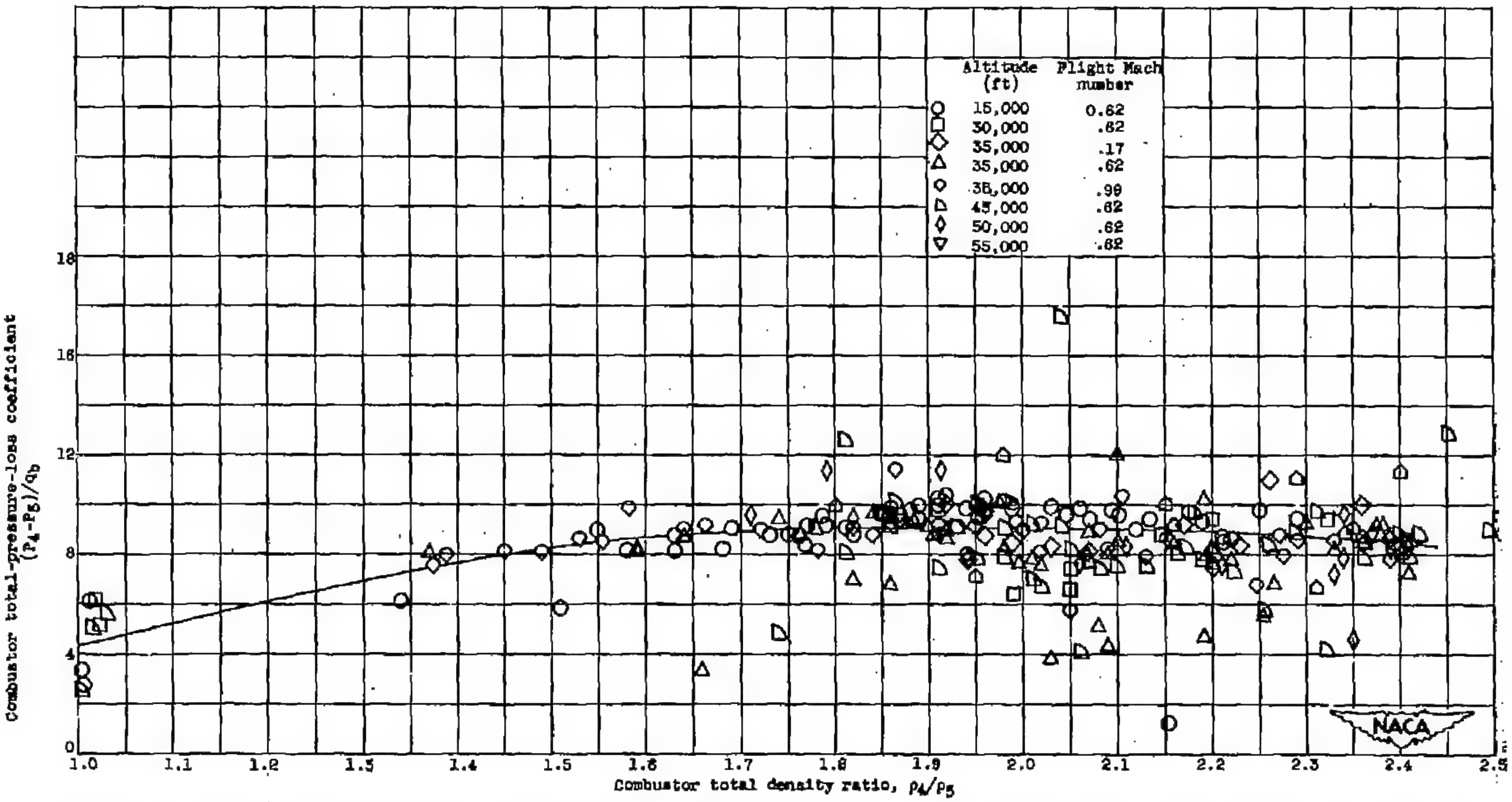
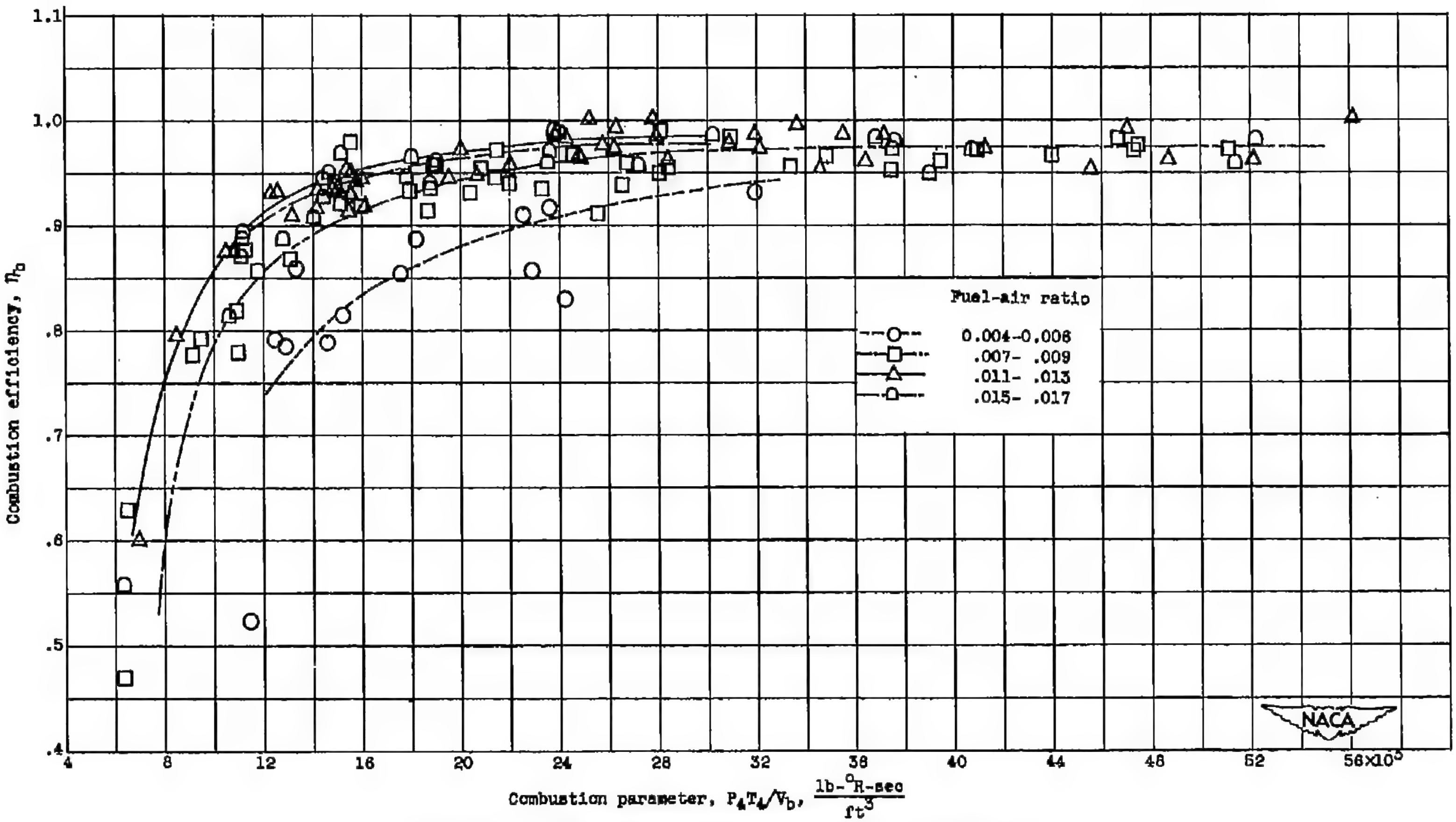


Figure 16. - Combustor pressure-loss characteristics in terms of combustor parameters. Prototype J40-WE-8 turbojet engine (compressor 1, combustor A).

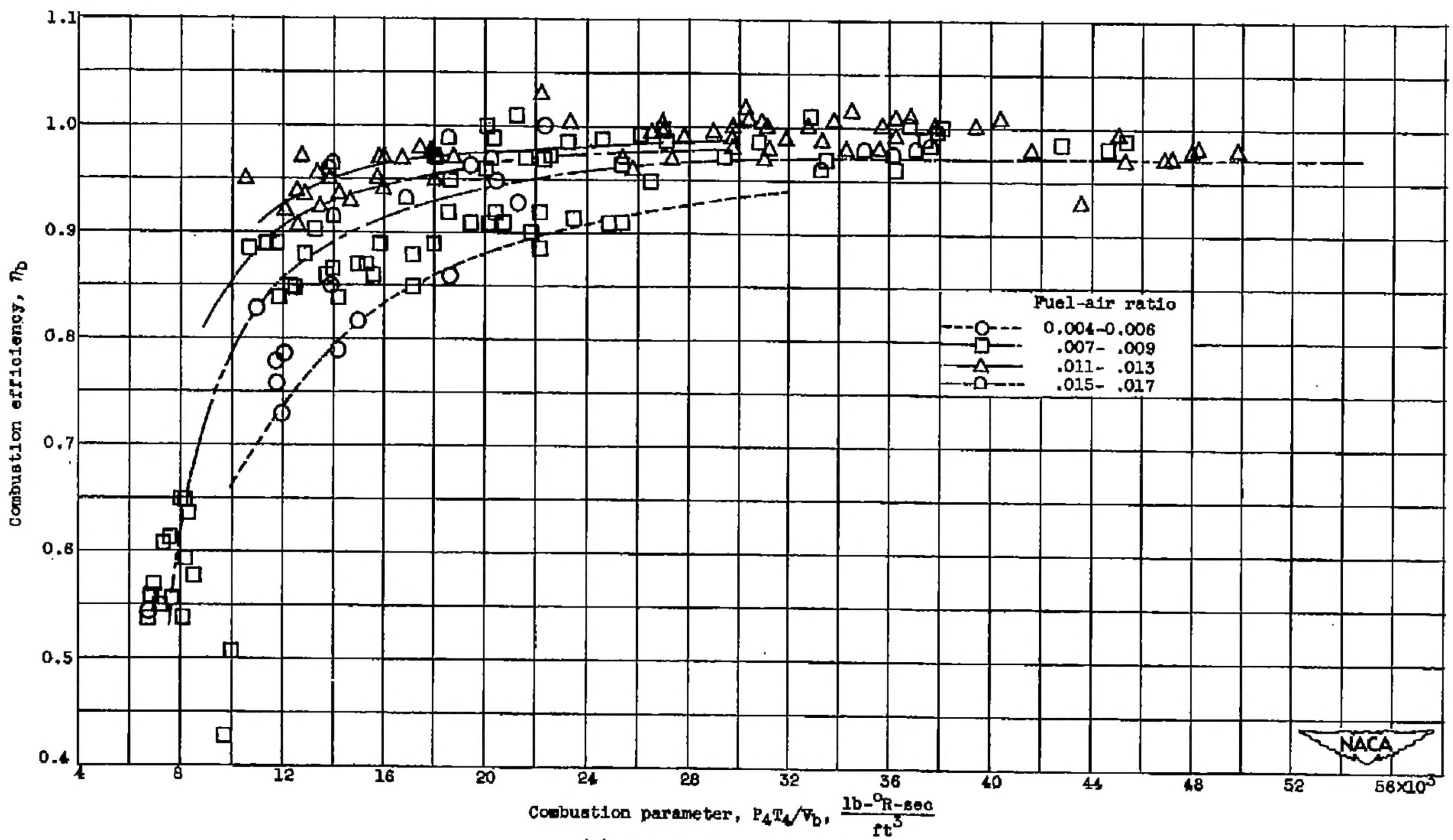
[redacted]



(a) Prototype J40-WE-8 turbojet engine (compressor 1, combustor A).

Figure 17. - Variation of combustion efficiency with combustion parameter.

[redacted]



(b) Compressor 2, combustor C.

Figure 17. - Concluded. Variation of combustion efficiency with combustion parameter.

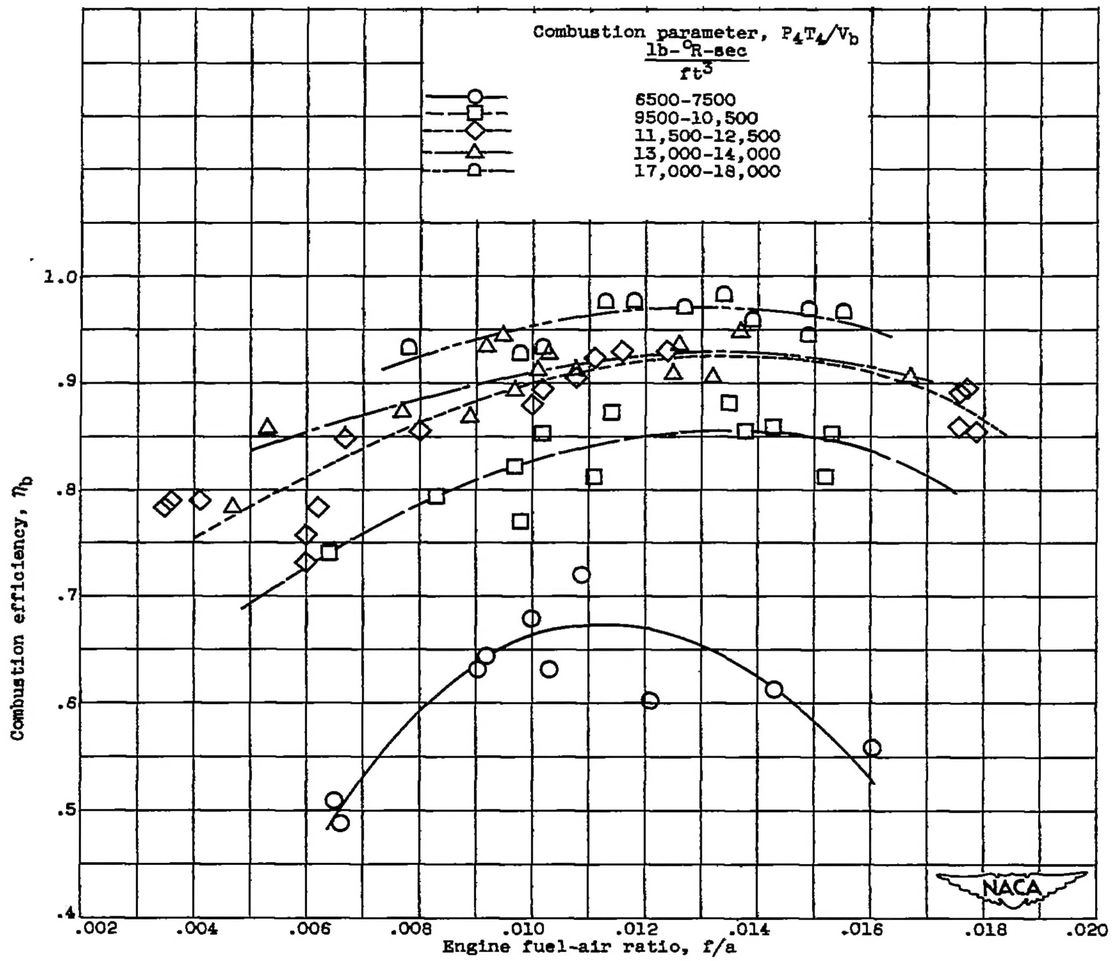
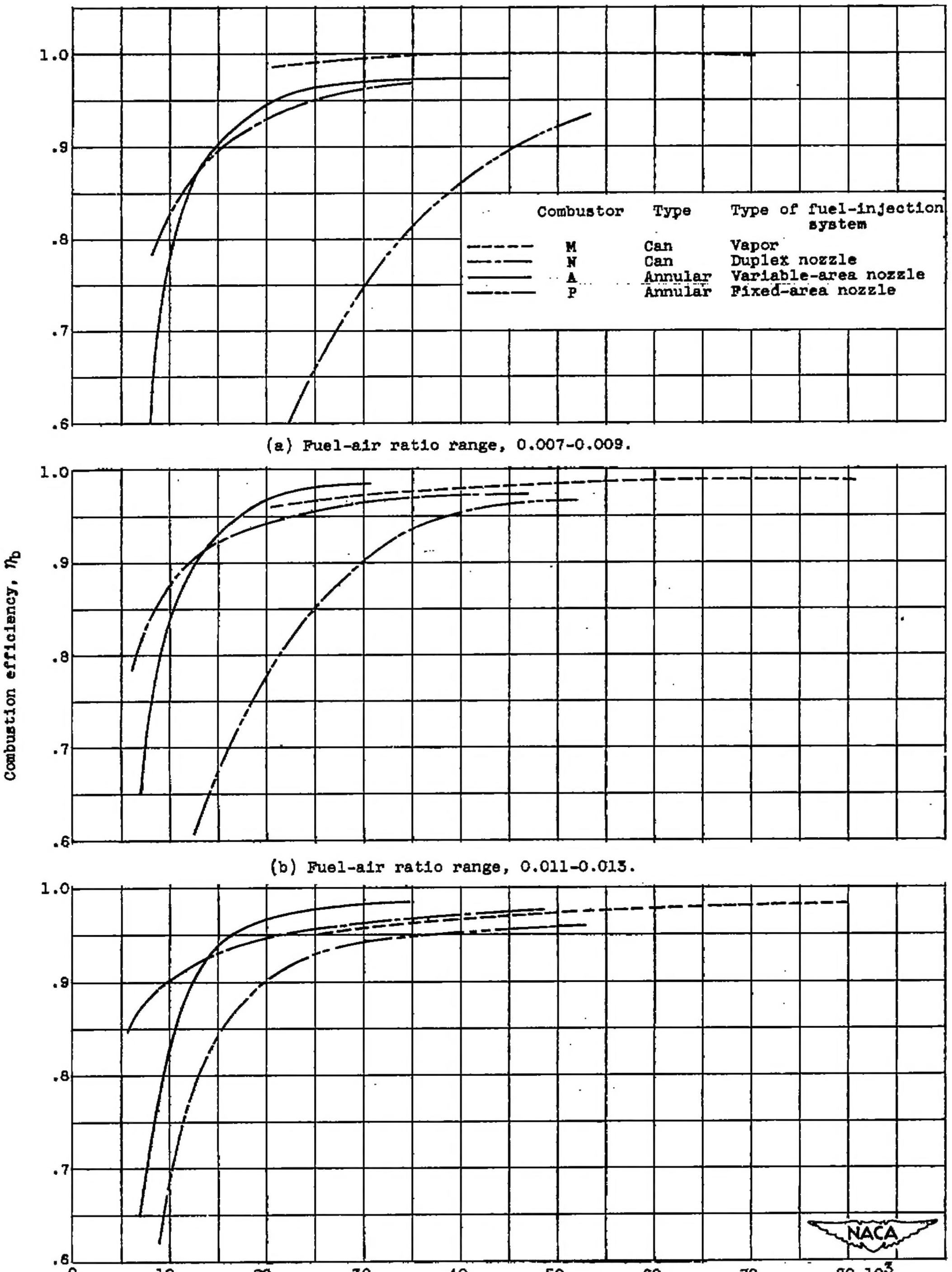


Figure 18. - Variation of combustion efficiency with fuel-air ratio for several values of combustion parameter. Compressors 1 and 2 with combustors A and B, respectively.



(a) Fuel-air ratio range, 0.007-0.009.

(b) Fuel-air ratio range, 0.011-0.013.

Combustion parameter, P_4T_4/V_b, $\frac{\text{lb-}^{\circ}\text{R-sec}}{\text{ft}^3}$

(c) Fuel-air ratio range, 0.015-0.017.

Figure 19. - Variation of combustion efficiency with combustion parameter for several unrelated combustors.

SECURITY INFORMATION

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